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( ) COMPARISON OF THEORETICALLY PREDICTED  
LATERAL-DIRECTIONAL AERODYNAMIC  
CHARACTERISTICS WITH FULL-SCALE WIND TUNNEL  
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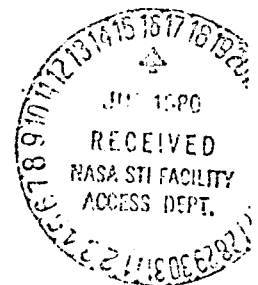
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COMPARISON OF THEORETICALLY PREDICTED  
LATERAL-DIRECTIONAL AERODYNAMIC  
CHARACTERISTICS WITH FULL-SCALE  
WIND TUNNEL DATA ON THE ATLIT  
AIRPLANE

KU-FRL-399-2

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# ABSTRACT

In this report an analytical method is presented for predicting lateral-directional aerodynamic characteristics of light twin-engine propeller-driven airplanes.

This method is applied to the Advanced Technology Light Twin-Engine (ATLIT) airplane. The calculated characteristics are correlated against full-scale wind tunnel data.

The method predicts the sideslip derivatives fairly well, although angle of attack variations are not well predicted. Spoiler performance was predicted somewhat high but was still reasonable. The rudder derivatives were not well predicted, in particular the effect of angle of attack. The predicted dynamic derivatives could not be correlated due to lack of experimental data.

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# LIST OF SYMBOLS

<u>Symbol</u>	<u>Definition</u>	<u>Dimension</u>
A	Aspect ratio	
$A_{v_{eff}}$	Effective vertical tail aspect ratio	
$B_1$	$1 - M^2$ ; compressibility factor	
$B_2$	$1 - \cos^2 \Lambda_c/4 M^2$ ; swept compressibility factor	
b	Span	m (ft)
$b_p$	Propeller blade width	m (ft)
$C_1, C_2$	Propeller sidewash factors	
$(C_{D_0})_w$	Wing zero lift drag coefficient	
$C_L$	Lift coefficient	
$C_{L_{max}}$	Maximum lift coefficient	
$C_{L_\alpha}$	Lift curve slope	deg <sup>-1</sup> , rad <sup>-1</sup>
$(C_{L_\alpha})_{N_p}$	Lift curve slope of the lift due to propeller normal force	deg <sup>-1</sup> , rad <sup>-1</sup>
$(C_{L_\alpha}')_v$	Effective $C_{L_\alpha}$ of the vertical tail	deg <sup>-1</sup> , rad <sup>-1</sup>
$C_{l_\alpha}$	Section lift curve slope	deg <sup>-1</sup> , rad <sup>-1</sup>
$\frac{C_{l_s}}{(C_{l_s})_{theory}}$	Empirical lift effectiveness correction factor	
$(C_{l_s})_{theory}$	Theoretical section lift effectiveness	deg <sup>-1</sup> , rad <sup>-1</sup>
$C_{l_{s_r}}$	Section lift effectiveness of rudder	deg <sup>-1</sup> , rad <sup>-1</sup>



# LIST OF SYMBOLS (continued)

<u>Symbol</u>	<u>Definition</u>	<u>Dimension</u>
$C_l$	Rolling moment coefficient	
$C_{l_p}$	Rolling moment due to roll rate	
$C_{l_r}$	Rolling moment due to yaw rate	
$C_{l_\beta}$	Rolling moment due to sideslip	$\text{deg}^{-1}, \text{rad}^{-1}$
$C_{l_s}$	Rolling moment effectiveness	$\text{rad}^{-1}$
$C_{l_{\delta_r}}$	Rolling moment due to rudder deflection	$\text{deg}^{-1}, \text{rad}^{-1}$
$C_{l_{\delta_s}}$	Rolling moment due to spoiler deflection	$\text{deg}^{-1}, \text{rad}^{-1}$
$(C_{N_a})_p$	Variation of propeller normal force with angle of attack	$\text{deg}^{-1}, \text{rad}^{-1}$
$C_n$	Yawing moment	
$C_{n_p}$	Yawing moment due to roll rate	$\text{rad}^{-1}$
$C_{n_r}$	Yawing moment due to yaw rate	$\text{rad}^{-1}$
$C_{n_\beta}$	Yawing moment due to sideslip	$\text{deg}^{-1}, \text{rad}^{-1}$
$C_{n_{\delta_r}}$	Yawing moment due to rudder deflection	$\text{deg}^{-1}, \text{rad}^{-1}$
$C_{n_{\delta_s}}$	Yawing moment due to spoiler deflection	$\text{deg}^{-1}, \text{rad}^{-1}$
$C_{Y_\beta}$	Side force due to sideslip	$\text{deg}^{-1}, \text{rad}^{-1}$
$C_{Y_{\delta_r}}$	Side force due to rudder deflection	$\text{deg}^{-1}, \text{rad}^{-1}$

# LIST OF SYMBOLS (continued)

<u>Symbol</u>	<u>Definition</u>	<u>Dimension</u>
$\bar{c}$	Mean aerodynamic chord	m (ft)
$c_1, c_2$	Wing taper factors	
$c_r$	Root chord	m (ft)
$c_t$	Tip chord	m (ft)
$(d_f)_h$	Width of the fuselage at the appropriate surface	m (ft)
$(d_f)_v$		
$(d_f)_w$		
$d_n$	Maximum depth of the nacelle forward of the wing	m (ft)
$f$	Propeller inflow factor	
$h$	Maximum fuselage height	m (ft)
$h_1$	Fuselage height at one-quarter of the fuselage length from the nose	m (ft)
$h_2$	Fuselage height at three-quarters of the fuselage length from the nose	m (ft)
$h_s$	Distance from airfoil mean line to end of spoiler	m (ft)
$J$	Viscous drag parameter	
$K'$	Correction factor for lift effectiveness at high rudder deflection angles	
$K_b$	Rudder span factor	
$K_h$	Relative tail size factor	
$K_i$	Wing-body interference factor	
$K_{\gamma}$	Compressibility correction for $C_{L_0}$	

# LIST OF SYMBOLS (continued)

<u>Symbol</u>	<u>Definition</u>	<u>Dimension</u>
$K_N$	1) Empirical factor for fuselage $C_{n\beta}$ in the presence of the wing (Section 4.2.3)  2) Empirical normal propeller force factor (Section 6.1.1)	
$k$	Empirical sweep correction factor for viscous drag	
$k_1, k_2$	Nacelle reduced mass factors	
$k_1'$	Relative body size factor	
$k_v$	$(C_{l_\alpha})_v / 2\pi$	
$l_f$	length of the fuselage	m (ft)
$l_{n_{eff}}$	length of the nacelle ahead of the wing	m (ft)
$l_v$	Distance from the center of gravity to the quarter-chord of the mean aerodynamic chord of the vertical tail, parallel to the X-body axis	m (ft)
$l_v'$	Distance parallel to the X-body axis from the center of gravity to the quarter chord of the mean aerodynamic chord of that portion of the vertical tail spanned by the rudder	m (ft)
$M$	Mach number	
$N_{Re}$	Reynolds number	
$n_n$	Number of nacelles	
prop	Propeller	
$\frac{q_v}{q_\infty}$	Dynamic pressure ratio at the vertical tail	

# LIST OF SYMBOLS (continued)

<u>Symbol</u>	<u>Definition</u>	<u>Dimension</u>
$\frac{\bar{q}_n}{\bar{q}_\infty}$	Dynamic pressure ratio at the nacelle	
$\bar{q}_\infty$	Free stream dynamic pressure	N/m <sup>2</sup> (lb/ft <sup>2</sup> )
$R_p$	Propeller radius	m (ft)
$S$	Area	m <sup>2</sup> (ft <sup>2</sup> )
$(S_f)_s$	Fuselage side area	m <sup>2</sup> (ft <sup>2</sup> )
$S_p$	Propeller disc area	m <sup>2</sup> (ft <sup>2</sup> )
$S_{x_n}$	Nacelle effective cross section area equal to a circular cross section with a diameter equal to $d_n$	m <sup>2</sup> (ft <sup>2</sup> )
$T$	Total propeller thrust	N (lb)
$T'_c$	$T/(\bar{q}_\infty S_w)$	
$t/c$	Maximum thickness ratio	
$\bar{v}$	Fuselage volume	m <sup>3</sup> (ft <sup>3</sup> )
$w_{\max}$	Maximum fuselage width	m (ft)
$\bar{x}$	$\bar{x}_{a.c.} - \bar{x}_{c.g.}$	
$x_{a.c.}$	Distance from the lifting surface apex to the aerodynamic center	m (ft)
$x_{ac_h} (c_{r_v}) / l_e$	Distance from leading edge of the vertical tail to the a.c. of the horizontal tail, in plane of the horizontal tail	m (ft)
$x_{c.g.}$	Distance from the center of gravity to the leading edge of the wing mean aerodynamic chord	m (ft)
$x_{c_r_h} (c_{r_v})$	Distance from root chord of vertical tail to root chord of horizontal tail	m (ft)

LIST OF SYMBOLS (continued)

<u>Symbol</u>	<u>Definition</u>	<u>Dimension</u>
$x_m$	Distance from the airplane nose to the center of gravity	m (ft)
$x_n$	Longitudinal distance from the center of gravity to the effective nacelle center of pressure	m (ft)
$x_p$	Longitudinal distance from the propeller to the airplane center of gravity	m (ft)
$x'_p$	Distance from the propeller to the wing quarter-chord along the thrust line	m (ft)
$x_s$	Distance from the airfoil leading edge to the end of the spoiler	m (ft)
$y^*$	Spanwise location of centroid of angle of attack loading	
$y_c$	Lateral distance from the root chord to the mean aerodynamic chord	m (ft)
$y_T$	Lateral distance from the thrust line to the center of gravity	m (ft)
$z_{c_{r_h}} (c_{r_v})$	Distance from the root chord of the vertical tail to the root chord of the horizontal tail	m (ft)
$z_n$	Vertical distance from the effective nacelle center of pressure to the center of gravity	m (ft)
$z_p$	Vertical distance from the thrust line to the airplane center of gravity	m (ft)
$z_v$	Distance from the center of gravity to the vertical tail mean aerodynamic chord, perpendicular to the X-body axis	m (ft)
$z'_v$	Distance parallel to the Z-body axis from the center of gravity to the quarter-chord of the mean aerodynamic chord of the portion of the vertical tail spanned by the rudder	m (ft)
$z_w$	Vertical distance from the X-axis of the equivalent circular fuselage to the quarter chord of the root chord of the exposed wing	m (ft)

# LIST OF SYMBOLS (continued)

<u>Symbol</u>	<u>Definition</u>	<u>Dimension</u>
$z'_w$	Vertical distance from the airplane to the quarter chord of the root chord of the exposed wing	m (ft)
<u>Greek Symbols</u>		
$\alpha$	Angle of attack relative to X-body axis	deg
$\alpha_{abs}$	Absolute angle of attack $\alpha - \alpha_0 + i$	deg
$\alpha_{0f}$	Fuselage zero lift angle of attack	deg
$(\alpha_\delta)_{C_L}$	$(\partial C_L / \partial \delta) / (\partial C_L / \partial \alpha)$	
$(\alpha_\delta)_{C_l}$	$(\partial C_l / \partial \delta) / (\partial C_l / \partial \alpha)$	
$\beta$	Sideslip angle	deg
$\beta'$	Propeller blade angle	deg
$\Gamma$	Dihedral angle	deg
$\Delta$	Empirical viscous drag increment	
$(\Delta C_L)_{N_p}$	Increment in lift coefficient due to propeller normal force	
$(\Delta C_L)_{\epsilon_p}$	Increment in lift coefficient due to propeller downwash	
$(\Delta C_{L_\alpha})_{w(\Delta q)}$	Increment in wing lift curve slope due to power induced increased dynamic pressure	deg <sup>-1</sup> , rad <sup>-1</sup>
$(\Delta C_{L_\alpha})_{w(\epsilon_p)}$	Increment in wing lift curve slope due to propeller downwash	deg <sup>-1</sup> , rad <sup>-1</sup>
$(\Delta C_{l_p})_{drag}$	Increment in $C_{l_p}$ due to viscous drag	rad <sup>-1</sup>
$(\Delta C_{l_p})_{N_p}$	Increment in $C_{l_p}$ due to propeller normal force	rad <sup>-1</sup>

# LIST OF SYMBOLS (continued)

<u>Symbol</u>	<u>Definition</u>	<u>Dimension</u>
$(\Delta C_{l_p})_{\text{power}}$	Increment in $C_{l_p}$ due to power	$\text{rad}^{-1}$
$(\Delta C_{l_p})_{w(\Delta \bar{q} + \epsilon_p)}$	Increment in wing contribution to $C_{l_p}$ due to increased dynamic pressure and propeller downwash	$\text{rad}^{-1}$
$(\Delta C_{l_\beta})_{N_p}$	Increment in $C_{l_\beta}$ due to propeller normal force	$\text{deg}^{-1}, \text{rad}^{-1}$
$(\Delta C_{l_\beta})_{n(\Delta \bar{q} + \epsilon_p)}$	Increment in nacelle contribution to $C_{l_\beta}$ due to increased dynamic pressure and propeller downwash	$\text{deg}^{-1}, \text{rad}^{-1}$
$(\Delta C_{l_\beta})_{\text{power}}$	Increment in $C_{l_\beta}$ due to power	$\text{deg}^{-1}, \text{rad}^{-1}$
$(\Delta C_{n_\beta})_{N_p}$	Increment in $C_{n_\beta}$ due to propeller normal force	$\text{deg}^{-1}, \text{rad}^{-1}$
$(\Delta C_{n_\beta})_{n(\Delta \bar{q})}$	Increment in nacelle contribution to $C_{n_\beta}$ due to increased dynamic pressure	$\text{deg}^{-1}, \text{rad}^{-1}$
$(\Delta C_{n_\beta})_{n(\sigma_p)}$	Increment in nacelle contribution to $C_{n_\beta}$ due to propeller sidewash	$\text{deg}^{-1}, \text{rad}^{-1}$
$(\Delta C_{n_\beta})_{\text{power}}$	Increment in $C_{n_\beta}$ due to power	$\text{deg}^{-1}, \text{rad}^{-1}$
$(\Delta C_{Y_\beta})_{N_p}$	Increment in $C_{Y_\beta}$ due to propeller normal force	$\text{deg}^{-1}, \text{rad}^{-1}$
$(\Delta C_{Y_\beta})_{n(\Delta \bar{q})}$	Increment in nacelle contribution to $C_{Y_\beta}$ due to increased dynamic pressure	$\text{deg}^{-1}, \text{rad}^{-1}$
$(\Delta C_{Y_\beta})_{n(\sigma_p)}$	Increment in nacelle contribution to $C_{Y_\beta}$ due to propeller sidewash	$\text{deg}^{-1}, \text{rad}^{-1}$

# LIST OF SYMBOLS (continued)

<u>Symbol</u>	<u>Definition</u>	<u>Dimension</u>
$(\Delta C_{Y\beta})_{\text{power}}$	Increment in $C_{Y\beta}$ due to power	$\text{deg}^{-1}, \text{rad}^{-1}$
$\frac{\Delta \bar{q}}{\bar{q}_\infty}$	Dynamic pressure increase behind the propeller as a ratio of the free stream dynamic pressure	
$\frac{\Delta \bar{q}_n}{\bar{q}_\infty}$	Dynamic pressure increase at the nacelles due to propeller as a ratio of the free stream dynamic pressure	
$\Delta \alpha$ 's	Spoiler lift effectiveness in terms of the change in zero-lift angle of attack	deg, rad
$\delta_R$	Rudder deflection	deg, rad
$\delta_s$	Spoiler deflection	deg, rad
$\frac{\partial C_{D_o}}{\partial \alpha}$	Variation of viscous drag with angle of attack	
$\frac{\partial \epsilon_p}{\partial \alpha_p}$	Downwash gradient behind the propeller	
$\frac{\partial \epsilon_u}{\partial \alpha_w}$	Upwash gradient at the propeller	
$\frac{\partial \sigma}{\partial \frac{pb_w}{2V}}$	Variation of sidewash with wing tip helix angle	
$\frac{\partial \sigma}{\partial \beta}$	Variation of sidewash with sideslip	
$\frac{\partial \sigma_p}{\partial \beta}$	Variation of propeller sidewash with sideslip	
$\epsilon_p$	Downwash angle behind the propeller	deg
$n$	Non-dimensional spanwise station, $y/(b/2)$	



# LIST OF SYMBOLS (continued)

<u>Symbol</u>	<u>Definition</u>	<u>Dimension</u>
$\theta$	1) Wing twist	deg
	2) Effective spoiler sweep angle	deg
$\Lambda$	Sweep angle	deg
$\Lambda_{\beta}$	Corrected sweep angle, $\tan^{-1}[\tan(\Lambda_{c/4})/\beta]$	deg
$\lambda$	Taper ratio	
$\phi_{TE}$	Section trailing-edge angle	deg
<u>Subscripts</u>		
c/2	Half-chord line	
c/4	Quarter-chord line	
c.g.	Center of gravity	
e	Exposed panel	
f	Fuselage	
fh	Fuselage-horizontal tail combination	
h	Horizontal tail	
/e	Leading edge	
M	High subsonic mach number	
max	Maximum	
n	Nacelle	
prop off	Propellers removed	
power on	Propellers installed and power on	
r	Root	
t	Tip	
v	Vertical tail	

LIST OF SYMBOLS (continued)

<u>Symbol</u>	<u>Definition</u>
w	Wing
wf	Wing-fuselage combination
wfh	Wing-fuselage-horizontal tail combination

## CHAPTER 1

### INTRODUCTION

This report describes work carried out under the second phase of a project performed by the Flight Research Laboratory of the University of Kansas sponsored by Grant MSG 1574 from the National Aeronautics and Space Administration (NASA), Langley Research Center. The purpose of this project was to correlate theoretically predicted aerodynamic characteristics of the Advanced Technology Light Twin (ATLIT) airplane with full-scale wind tunnel and flight test data. The original phases of this project were

1. correlate theoretical predictions of longitudinal aerodynamic characteristics with full-scale wind tunnel data;
2. correlate theoretical predictions of lateral-directional aerodynamic characteristics with full-scale wind tunnel data;
3. correlate the results of point 1 and point 2 with flight test data.

The results of the longitudinal correlation (point 1) are presented in Reference 1. This report deals only with the lateral-directional correlation (point 2). Full scale wind tunnel data used in the correlation were taken from Reference 2. The methods used in the prediction were taken from References 3 and 4.

The purpose of this report is to present an analytical method for predicting the lateral-directional aerodynamic characteristics of this class of airplane and to correlate the predictions with

full-scale wind tunnel tests to identify areas where current predictions fall short.

This report was originally to correlate the predicted characteristics with flight test data. However, aside from some performance data presented in Reference 5, sufficient flight test data is not yet available to perform a meaningful correlation. After consulting with NASA, it was agreed to delete the flight test correlation phase. Additional work of a similar nature (full-scale wind tunnel correlation) is presently being performed on another airplane in place of the ATLIT flight test correlation. Some derivatives for which no wind tunnel data were available are still presented for future comparison when this data becomes available.

## CHAPTER 2

### THE ATLIT AIRPLANE

The ATLIT airplane is a Piper PA-34-200 Seneca I general aviation low-wing monoplane with a number of high technology modifications. The major physical characteristics are listed in Table 2.1, and a three-view drawing is shown in Figure 2.1.

The improvements implemented on the ATLIT were

1. The original untapered, aspect ratio 7.25 wing having a 65<sub>2</sub>412 airfoil and an area of 19.4 m<sup>2</sup> (208.7 ft<sup>2</sup>) was replaced by a new tapered wing (taper ratio 0.5) with a higher aspect ratio (10.32) and a smaller area (14.4 m<sup>2</sup>). The new wing used a 17-percent thick GA(W)-1 airfoil.
2. Full span 30-percent chord Fowler flaps replaced the original 20-percent chord plain flaps.
3. Spoilers were installed for roll control instead of ailerons.

The new wing area was made smaller and the aspect ratio increased to provide better drag characteristics than the original wing. The GA(W)-1 airfoil was used for its high lift-to-drag ratio and high maximum lift coefficient. For acceptable landing speeds, full span flaps were used with spoilers providing roll control. During flight test, supercritical propellers were also evaluated. However, these were not used in the full-scale wind tunnel tests and are therefore not considered here.

In addition a number of changes were made in part of the wind tunnel test to clean up the drag of the ATLIT. This resulted in "as built" and "fully clean" wind tunnel tests. All comparisons in this report are done using the "fully clean" wind tunnel data.

Table 2.1: Specifications of the ATLIT Airplane

Wing:

Area, $m^2$ ( $ft^2$ )	14.40 (155.0)
Span, m (ft)	12.19 (40.0)
Aspect ratio	10.32
Thickness ratio	0.17
Dihedral, deg	7
Taper ratio	0.5
Incidence angle at root, deg	0.5
Incidence angle at tip, deg	-2.5
Leading edge and trailing edge sweep, deg	3.67
Mean aerodynamic chord, m (ft)	1.225 (4.018)
Root chord, m (in)	1.575 (62.0)
Tip chord, m (in)	0.787 (31.0)
Airfoil	GA(W)-1

Flap:

Type	Fowler
Area (total), $m^2$ ( $ft^2$ )	3.56 (38.3)
Span/side, m (ft)	5.15 (16.91)
Chord, perc. of wing chord	30
Inboard wing station, m (in)	0.711 (28.0)
Outboard wing station, m (in)	5.867 (231.0)
Leading edge retracted, perc. of wing chord	70
Maximum deflection, deg	40
Maximum travel, m (in)	0.343 (13.50)
Root chord, m (in)	0.445 (17.50)
Tip chord, m (in)	0.244 (9.62)

Table 2.1: Continued

Spoiler:

Type	Differential-Slat- Upper Surface
Area (total), $m^2$ ( $ft^2$ )	0.488 (5.25)
Span/side, m (in)	3.226 (127.0)
Inboard wing station, m (in)	2.565 (101.0)
Outboard wing station, m (in)	5.791 (228.0)
Hinge, perc. of wing chord	70
Maximum deflection, deg	60
Root chord, m (in)	0.177 (6.975)
Tip chord, m (in)	0.124 (4.880)

Tail:

Stabilator:

Area (including tab), $m^2$ ( $ft^2$ )	3.60 (38.7)
Span, m (ft)	4.13 (13.56)
Chord (constant), m (in)	0.371 (14.29)
Aspect ratio	4.75
Sweep angle, deg	0
Dihedral, deg	0
Hinge line, perc. of chord	26.6
Airfoil	NACA 0010

Vertical:

Area (including tab), $m^2$ ( $ft^2$ )	1.35 (19.9)
Span, m (ft)	1.52 (5.00)
Aspect ratio	1.17
Taper ratio	0.42
Root chord, m (in)	1.708 (67.23)
Tip chord, m (in)	0.723 (28.45)



Table 2.1: Continued

Mean aerodynamic chord, m (in)	1.232 (50.47)
Leading edge sweep angle, deg	39.92
Airfoil	NACA 0009
Stabilator tab:	
Area, m <sup>2</sup> (ft <sup>2</sup> )	0.539 (5.8)
Span, m (in)	3.023 (119.0)
Chord (constant), m (in)	0.178 (7.0)
Tab hinge line to stabilator hinge line, m (in)	0.462 (18.17)
Rudder:	
Area (including tab), m <sup>2</sup> (ft <sup>2</sup> )	0.706 (7.6)
Span, m (in)	1.525 (60.02)
Chord (constant), m (in)	0.449 (17.69)
Rudder trim tab:	
Area, m <sup>2</sup> (ft <sup>2</sup> )	0.158 (1.7)
Span, m (in)	0.895 (35.25)
Chord (constant), m (in)	0.175 (6.9)
<u>Powerplant:</u>	
Number of engines	2
Manufacturer	Lycoming
Model	IO-360-C1E6
Takeoff rpm	2700
Takeoff power, hp	200
<u>Propeller:</u>	
Manufacturer	Hartzell
Model	HC-C1YK-ICEF/FO 7640A
Number of blades	2
Diameter, m (in)	1.330 (52.0)

Table 2.1: Concluded

Weights and Loading:

Gross weight, N (lb)	17,792 (4,000)
Empty weight, N (lb)	11,018 (2,477)
Useful load, N (lb)	6,774 (1,523)
Wing loading (at gross weight), $N/m^2$ (lb/ft <sup>2</sup> )	1,236 (25.8)

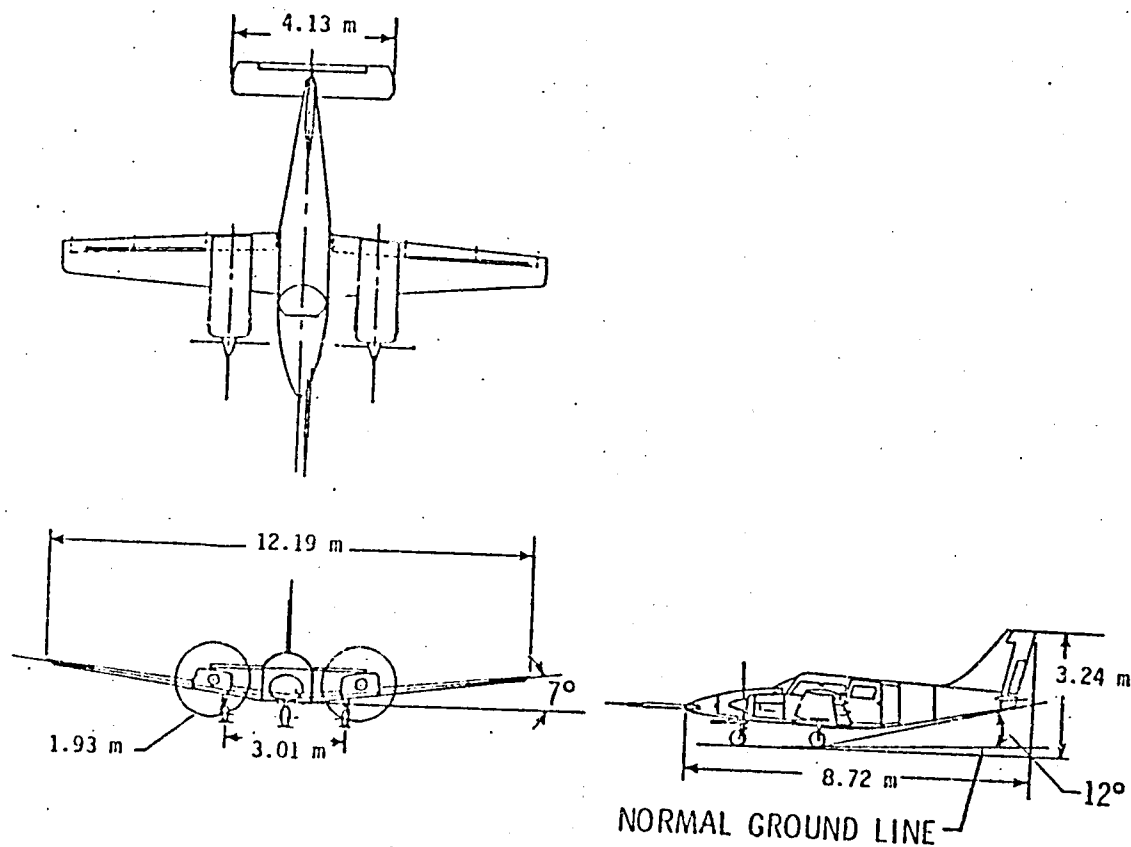


Figure 2.1: Three-view drawing of the ATLIT airplane (Reference 1)

## 2.1 Geometric Parameters of Wing and Tail

Depending on which characteristic has to be determined, the total area or the exposed area of the wing and the tail are considered. The total planform is considered to extend through the nacelle and the fuselage, while the exposed planform terminates at the fuselage. Pertinent dimensions for the wing, the horizontal tail and the vertical tail are shown in Figures 2.1.1, 2.1.2 and 2.1.3, respectively. Table 2.1.1 lists the geometric parameters of the wing and the tail pertinent in the analysis.

Table 2.1.1: Pertinent Wing and Tail Geometric Parameters Used in the Analysis

Symbol	Description	Wing		Horizontal Tail		Vertical Tail
		Total	Exposed	Total	Exposed	Exposed
S	Area, m <sup>2</sup> (ft <sup>2</sup> )	14.40 (155.0)	12.53 (134.8)	3.60 (38.7)	3.25 (34.9)	1.75 (18.8)
b	Span, m (ft)	12.19 (40.0)	10.96 (36.0)	4.13 (13.56)	3.73 (12.23)	1.52 (5.0)
A	Aspect ratio, b <sup>2</sup> /S	10.32	9.61	4.75	4.28	1.33
c <sub>t</sub>	Tip chord, m (in)	0.787 (31.00)	0.787 (31.00)	0.871 (34.29)	0.871 (34.29)	0.723 (28.45)
c <sub>r</sub>	Root chord, m (in)	1.575 (62.00)	1.495 (58.87)	0.871 (34.29)	0.871 (34.29)	1.575 (62.0)
λ	Taper ratio, c <sub>t</sub> /c <sub>r</sub>	0.50	0.527	1	1	0.459
$\bar{c}$	* Mean aerodynamic chord, m (in)	1.225 (48.22)	1.178 (46.38)	0.871 (34.29)	0.871 (34.29)	1.201 (47.30)
y <sub>c</sub>	** Lateral position of mean aerodynamic chord, m (in)	2.709 (106.67)	2.46 (96.85)	1.033 (40.68)	0.932 (36.69)	0.334 (13.15)
Γ	Dihedral angle, deg.	7	7	0	0	-
Λ <sub>le</sub>	Leading-edge sweep, deg.	3.67	3.67	0	0	40.0
Λ <sub>c/4</sub>	Sweep of c/4 line, deg.	1.835	1.835	0	0	34.5
Λ <sub>c/2</sub>	Sweep of c/2 line, deg.	0	0	0	0	29.0

$$* \bar{c} = \frac{2}{3} c_r \left( \frac{1 + \lambda + \lambda^2}{1 + \lambda} \right)$$

$$** y_c = \frac{1}{3} \left( \frac{1 + 2\lambda}{1 + \lambda} \right) \frac{b}{2}$$

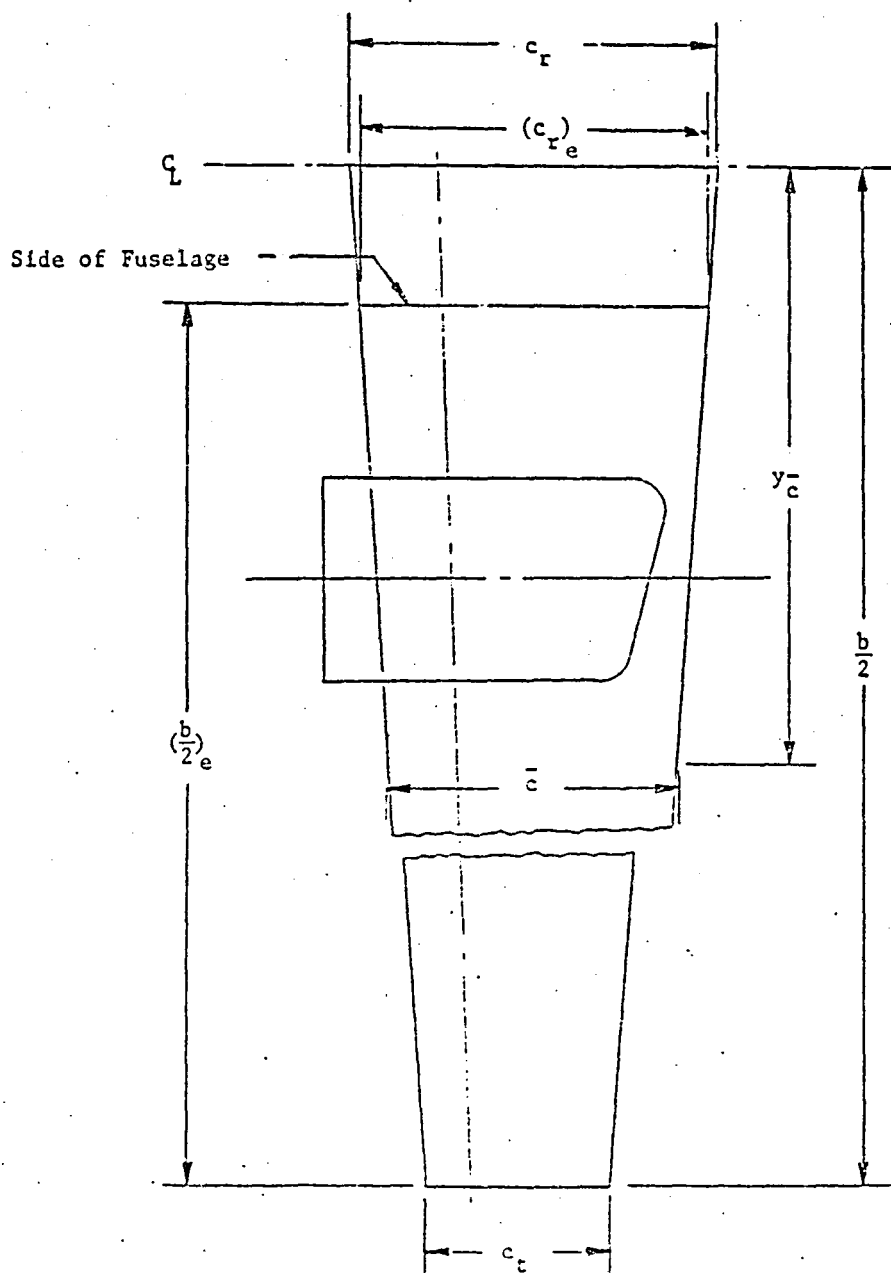


Figure 2.1.1: Definition sketch of wing dimensions

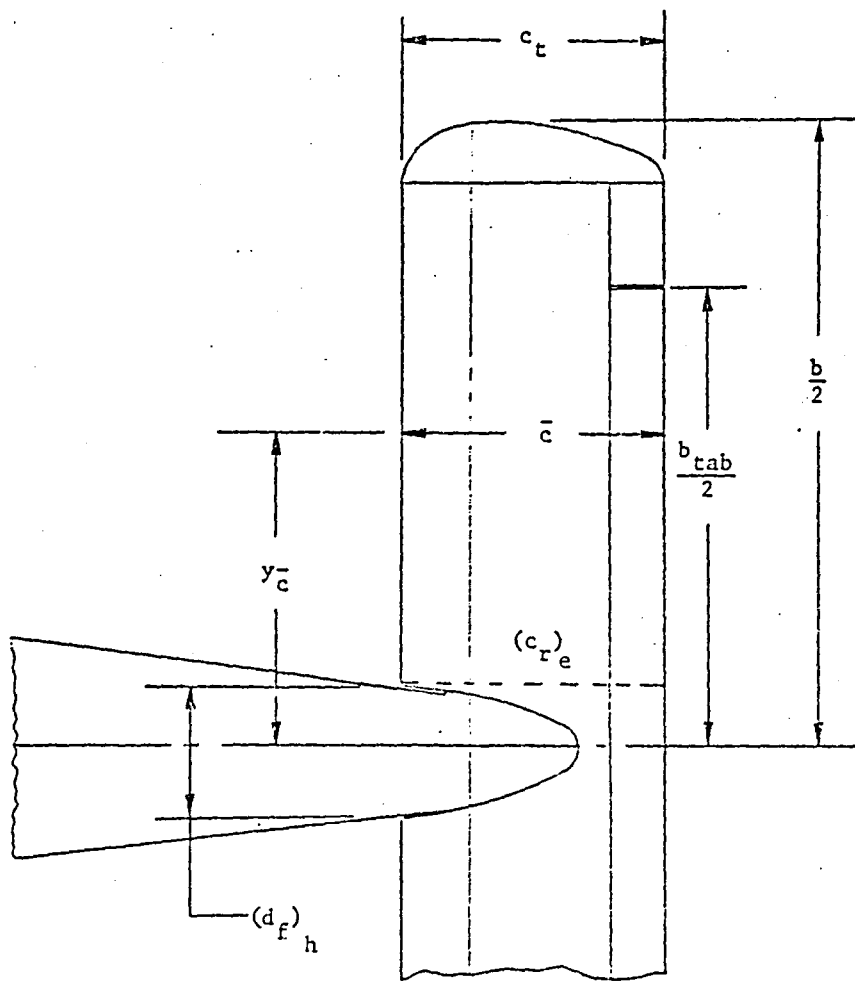


Figure 2.1.2: Definition sketch of horizontal tail dimensions

Figure 2.1.3: Definition sketch of vertical tail dimensions



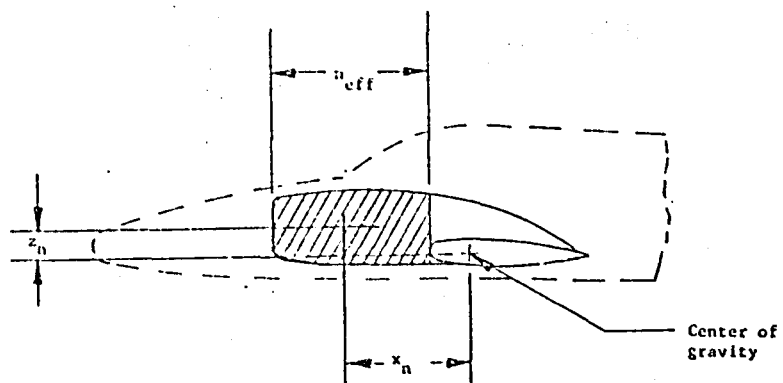
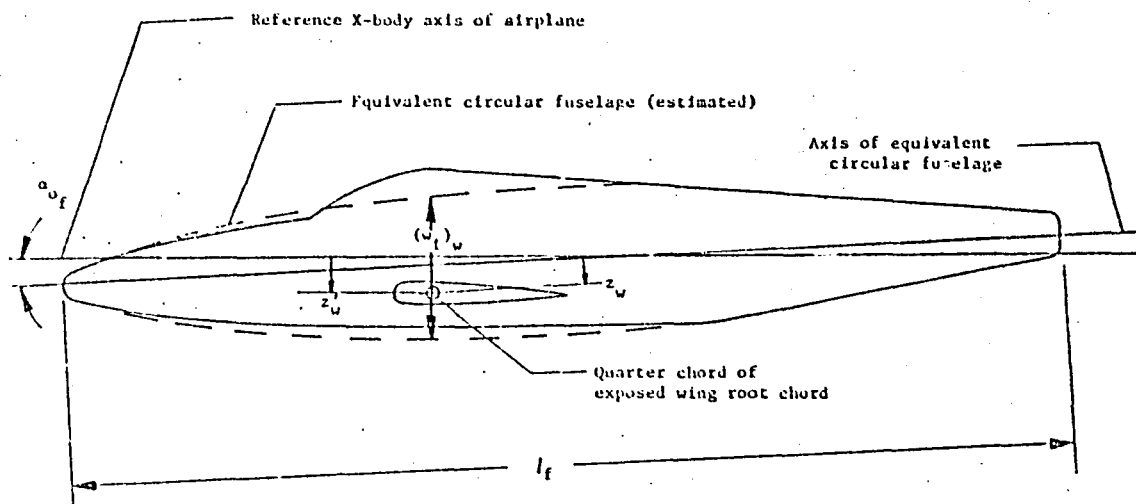


Figure 2.1.4: Definition sketch of the fuselage and nacelles

## CHAPTER 3

### PRESENTATION OF RESULTS

In this chapter, the predicted lateral-directional characteristics of the ATLIT airplane will be compared with full-scale wind tunnel data of Reference 2.

#### 3.1 Sideslip Derivatives, $C_{Y_\beta}$ , $C_{n_\beta}$ , $C_{l_\beta}$

The methods used in predicting the sideslip derivatives of the ATLIT airplane are presented in Sections 4.1 to 4.3. In Figures 3.1.1 to 3.1.6 the predicted derivatives  $C_{Y_\beta}$ ,  $C_{n_\beta}$ , and  $C_{l_\beta}$  are presented and compared with full-scale wind tunnel data for both propellers off and power-on.

Figures 3.1.1 and 3.1.2 show the propeller off and power-on  $C_{Y_\beta}$ . In general the results show good agreement, although the effect of angle of attack has not been properly accounted for--especially for the power-off case. It is felt that the angle of attack effect on the fuselage and fuselage-vertical tail combination has not been accounted for.

The  $C_{n_\beta}$  of the full-scale wind tunnel test shows a strong variation with angle of attack which the theoretical methods did not predict. While the average values agree fairly well, the predictions show a basically linear change in  $C_{n_\beta}$  with angle of attack. With power on, the prediction is even worse, as the wind tunnel  $C_{n_\beta}$  varied even more with angle of attack. As is discussed in Section 4.2.2, the wing position plays an important role in the fuselage contribution to  $C_{n_\beta}$ . However, there is not enough data available to account accurately for this factor.

Figures 3.1.5 and 3.1.6 show the propellers off and power on  $C_{2B}$  prediction compared with wind tunnel data. For the propellers off case the prediction is somewhat high, but the variation with angle of attack agrees with wind tunnel results. With power on, the wind tunnel tests show a more pronounced variation with angle of attack. At the low angles of attack the prediction is not very good, but it improves as the angle of attack is increased.

### 3.2 Control Derivatives

The control derivatives for the spoiler and the rudder are predicted in Sections 4.4 and 4.5. Figure 3.2.1 presents the spoiler effectiveness for  $\alpha = 0$  as a function of spoiler deflection. The predicted effectiveness is slightly higher than the wind tunnel data. Nevertheless the prediction is considered quite good, considering the general nature of the prediction method and the lack of data for this class of airplane.

Figure 3.2.2 presents the yawing moment due to spoiler deflection. Again the prediction is slightly higher than the wind tunnel data, although the overall trend is predicted quite well.

In Section 6.1.5 it is shown that the effect of power on the spoiler characteristics were significant in the wind tunnel tests; no method was found to predict this effect.

Figures 3.2.3 through 3.2.5 compare the predicted rudder derivatives with full-scale wind tunnel data. In general the predictions are not very good. The predictions show little variation with angle of attack (except, of course, for  $C_{L,R}$ ),

while the wind tunnel data exhibits considerable variation with angle of attack. There is little variation in the rudder derivatives with power in the wind tunnel data. It was assumed that there was no power effect on the predicted rudder derivatives.

### 3.3 Dynamic Derivatives

The dynamic derivatives of the ATLIT airplane were originally predicted for comparison with flight test data. This data is not yet available; however, the data is presented here for completeness. Power effects were only considered significant for  $C_{\ell p}$ . Figures 3.3.1 through 3.3.5 show the predicted derivatives as a function of angle of attack.

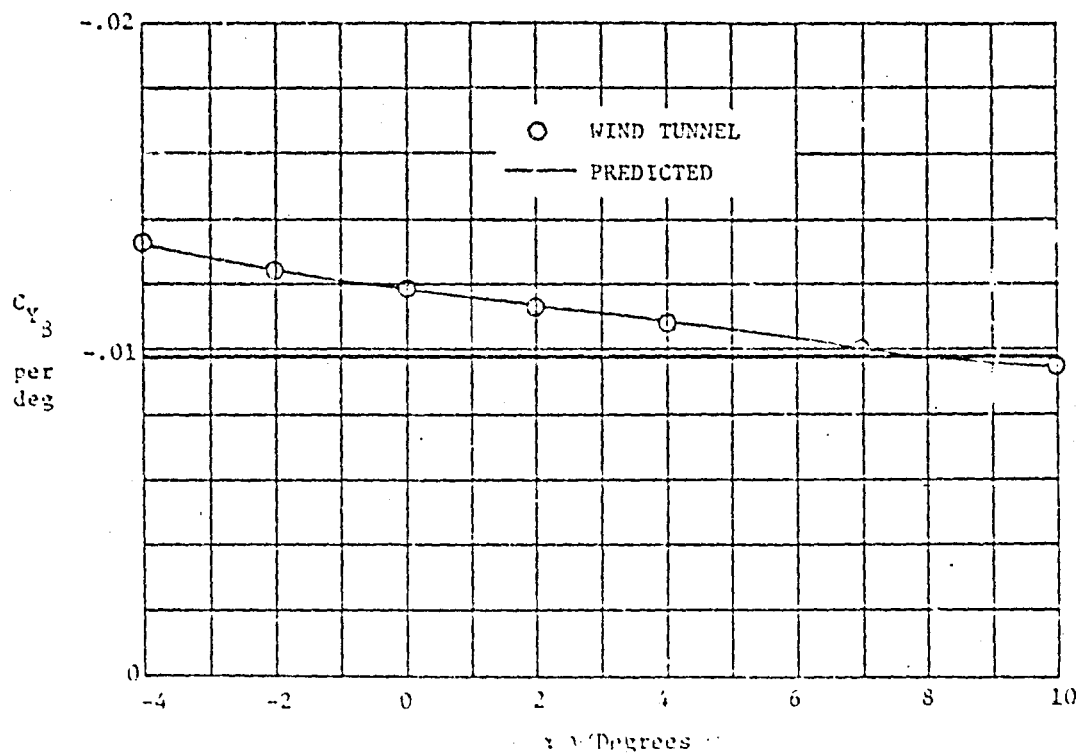


Figure 3.1.1: Comparison of predicted  $C_{D3}$  with full scale wind tunnel data (propellers removed,  $N_{Re} = 2.3$  million)

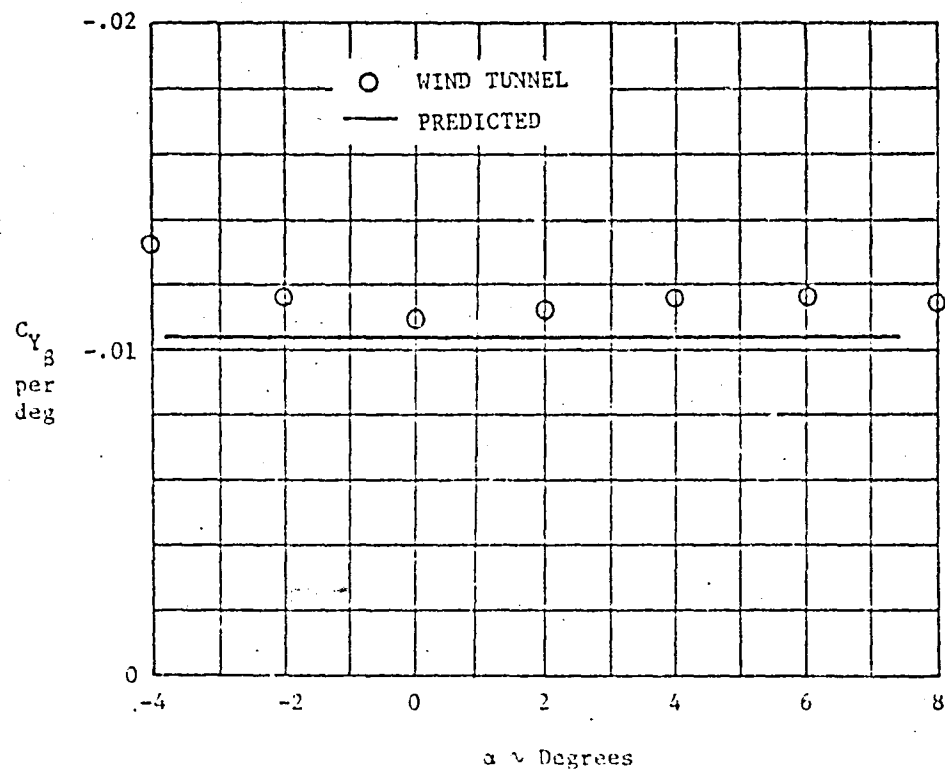


Figure 3.1.2: Comparison of predicted power-on  $C_{Y_3}$  with wind tunnel data ( $T'_c = .1970$ )

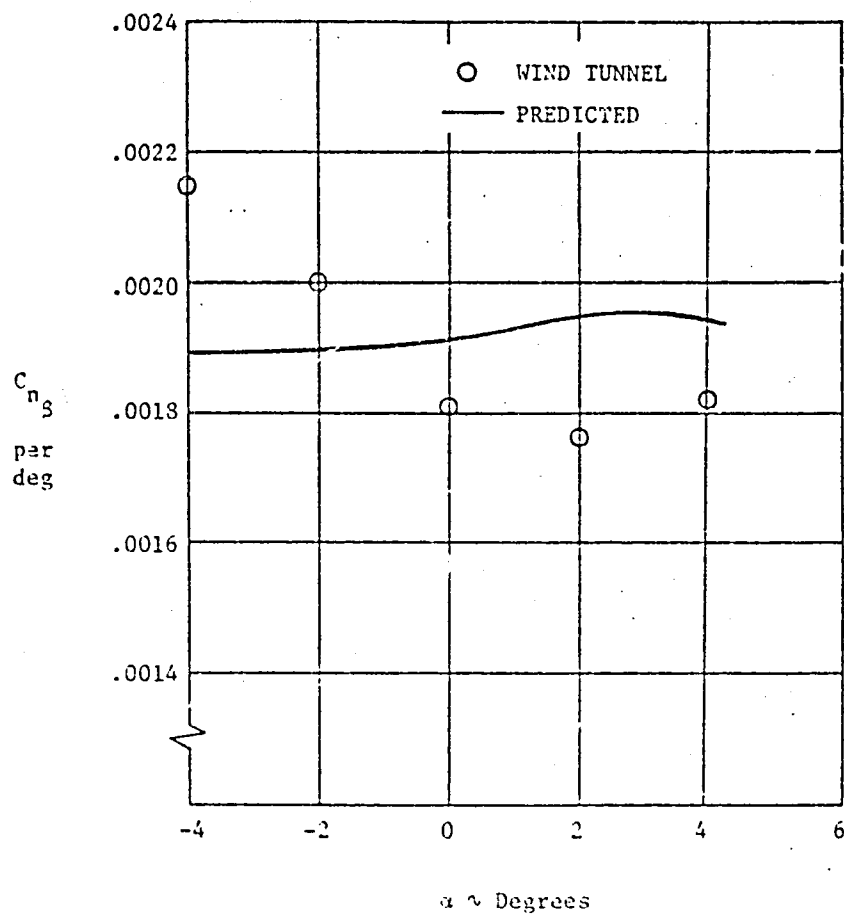


Figure 3.1.3: Comparison of predicted  $C_{n3}$  with full scale wind tunnel data (propellers removed,  $N_{Re} = 1.3$  million)

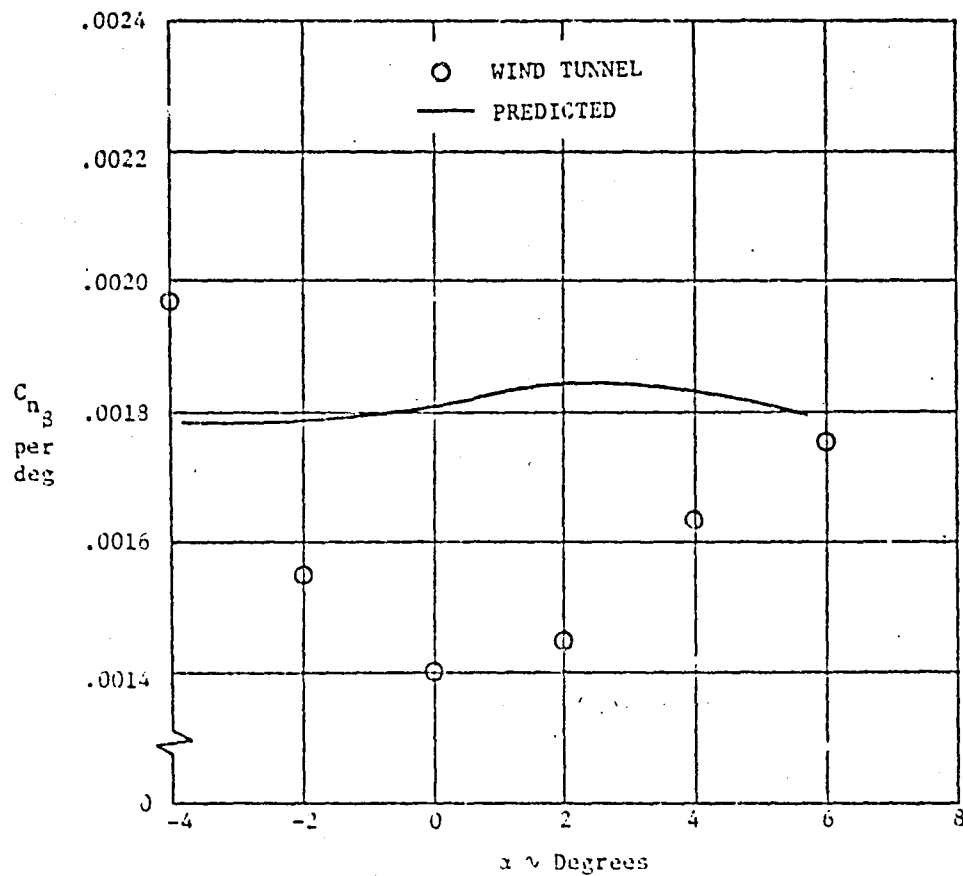


Figure 3.1.4: Comparison of predicted power-on  $C_{n_2}$  with wind tunnel data on the ATLIT Airplane ( $T'_c = .1970$ )



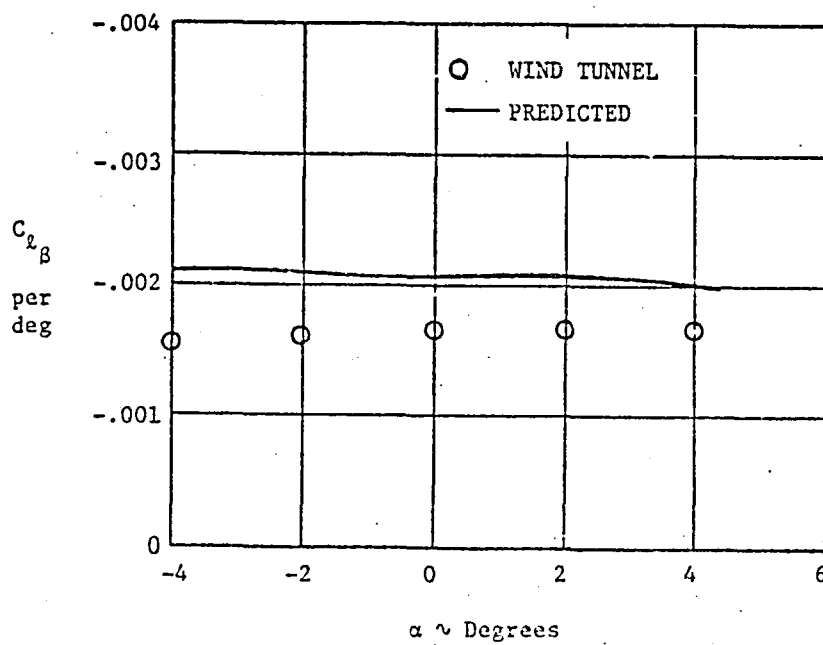


Figure 3.1.5: Comparison of predicted  $C_{l_B}$  with full scale wind tunnel data (propellers removed,  $N_{Re} = 2.3$  million)

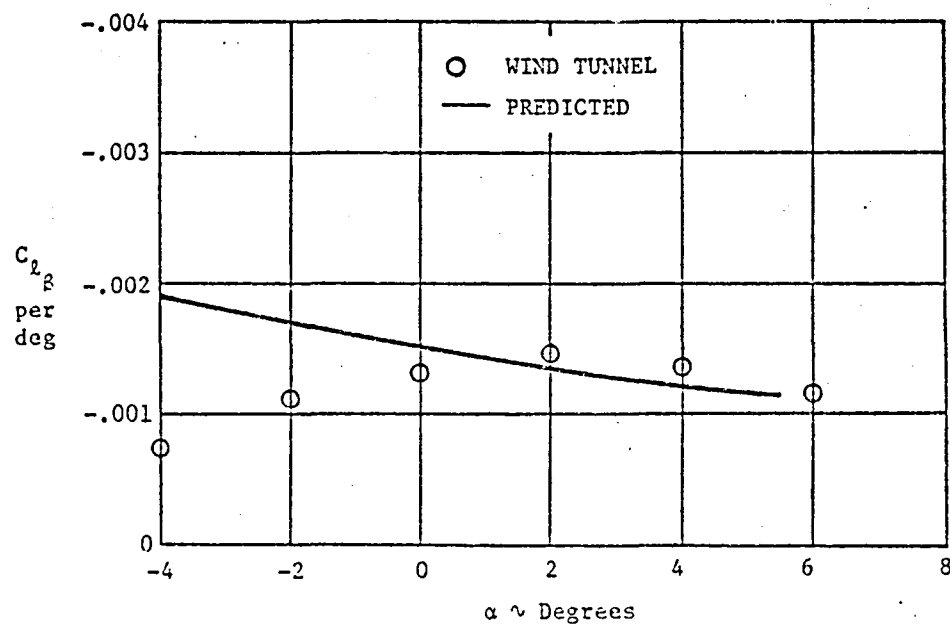


Figure 3.1.6: Comparison of predicted power-on  $C_{L_{\beta}}$  with wind tunnel data on the ATLIT Airplane ( $T'_C = .1970$ )

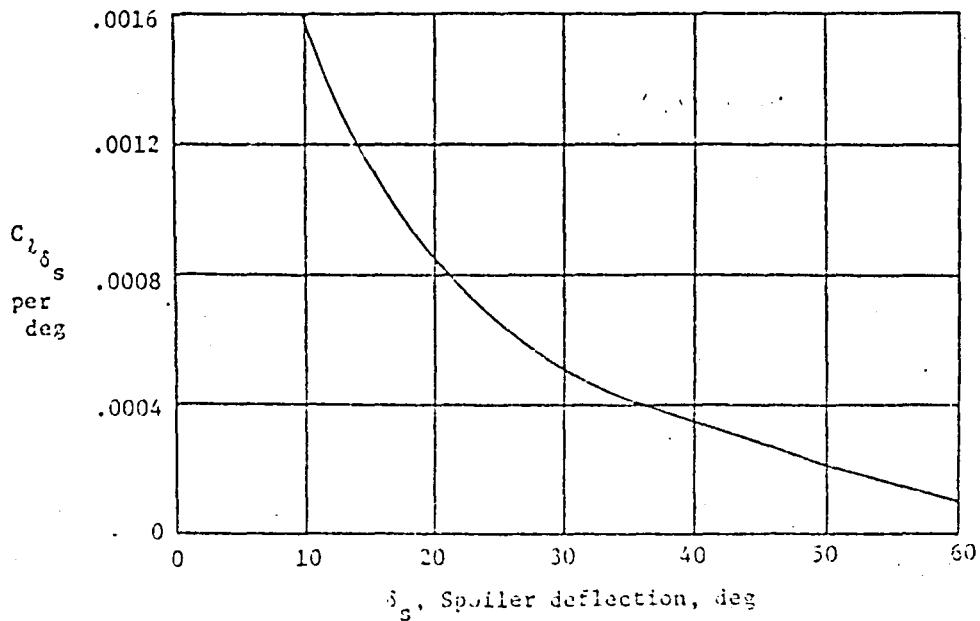
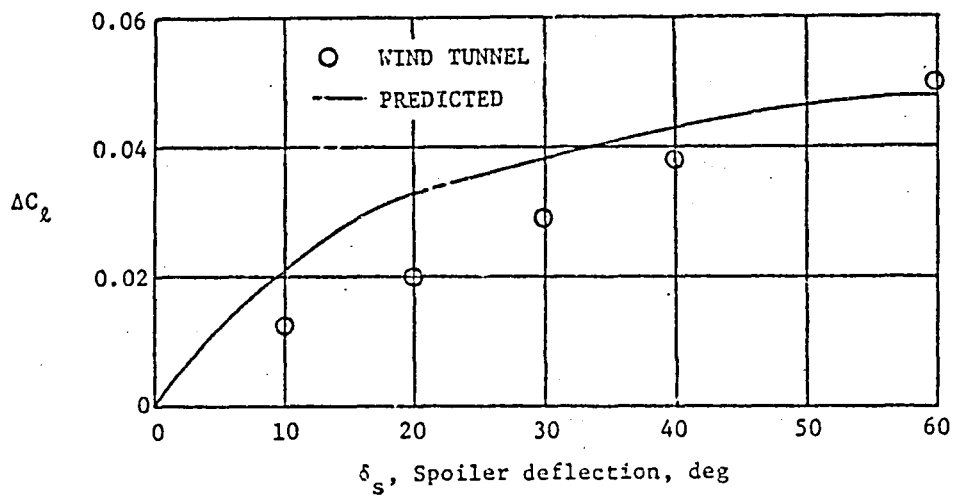


Figure 3.2.1: Comparison of predicted spoiler effectiveness with full-scale wind tunnel data (Power off;  $\alpha = 0^\circ$ )

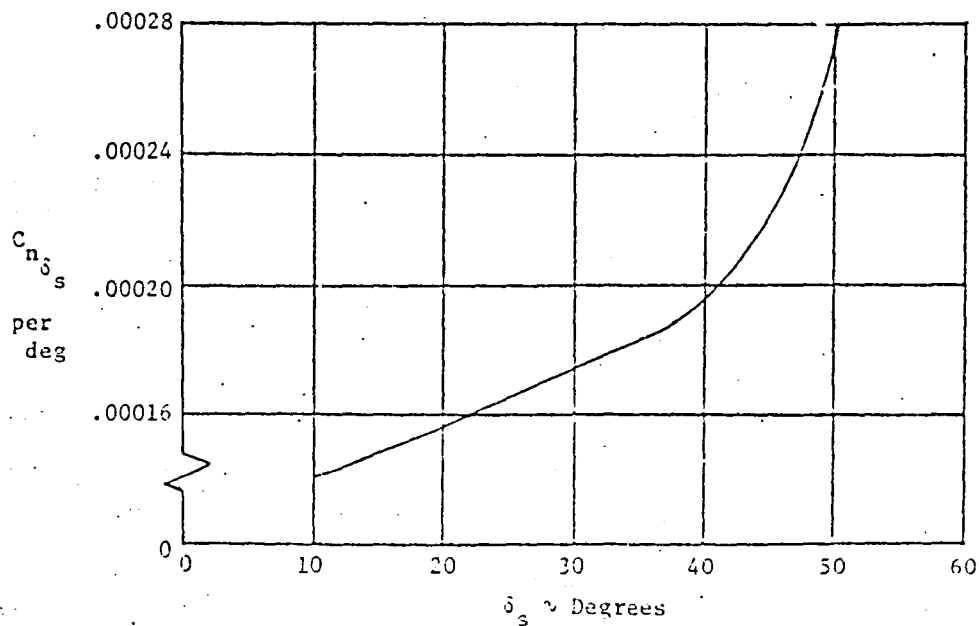
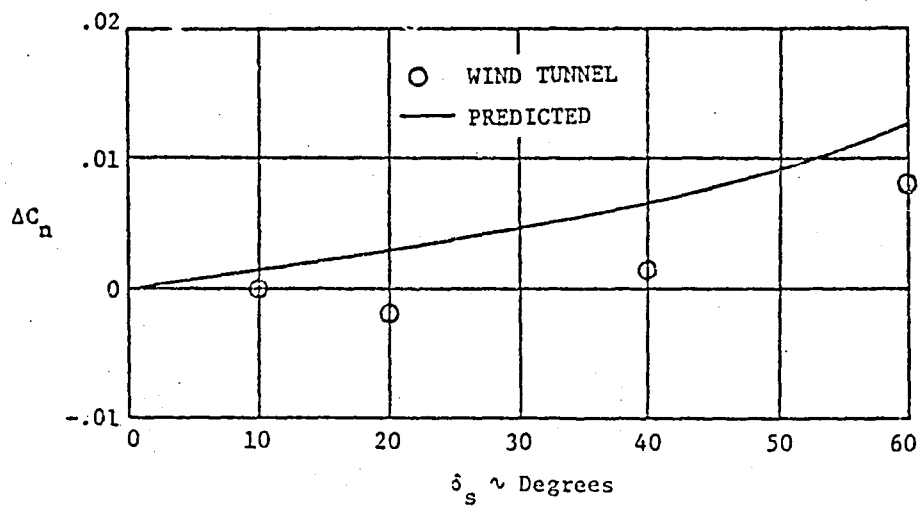


Figure 3.2.2: Comparison of predicted yaw due to spoilers with wind tunnel data (propellers removed,  $\alpha=0^\circ$ ).

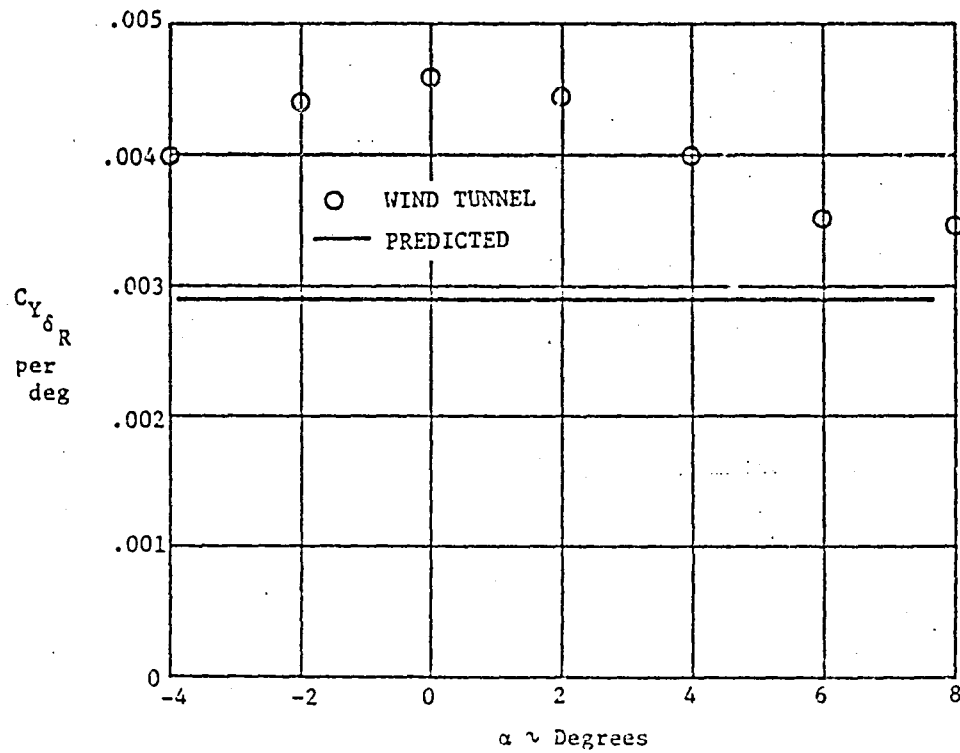


Figure 3.2.3: Comparison of predicted  $C_{Y\delta_R}$  with wind tunnel data  
 ( $\delta_R = \pm 10^\circ$ ,  $\delta = 0^\circ$ )

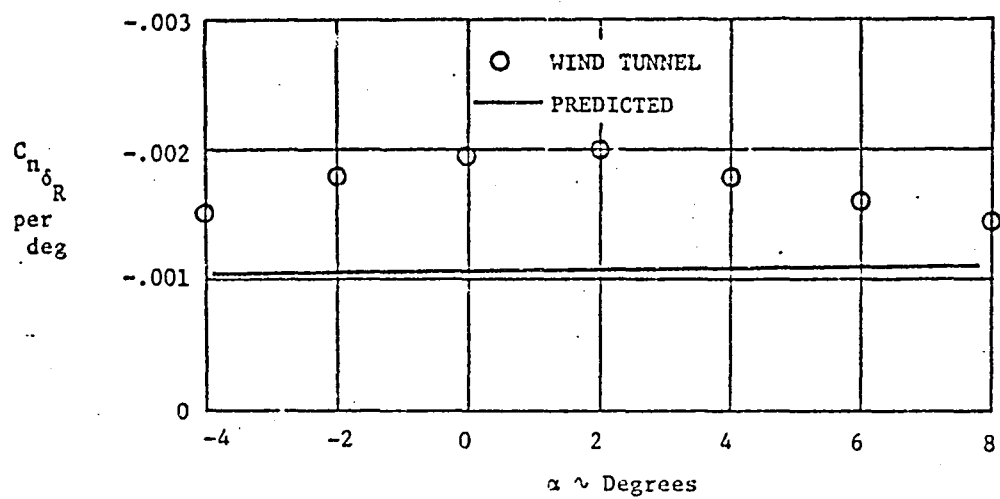


Figure 3.2.4: Comparison of predicted  $C_{n\delta_R}$  with wind tunnel data  
 ( $\beta = 0^\circ$ ,  $\delta_R = -10^\circ$ )

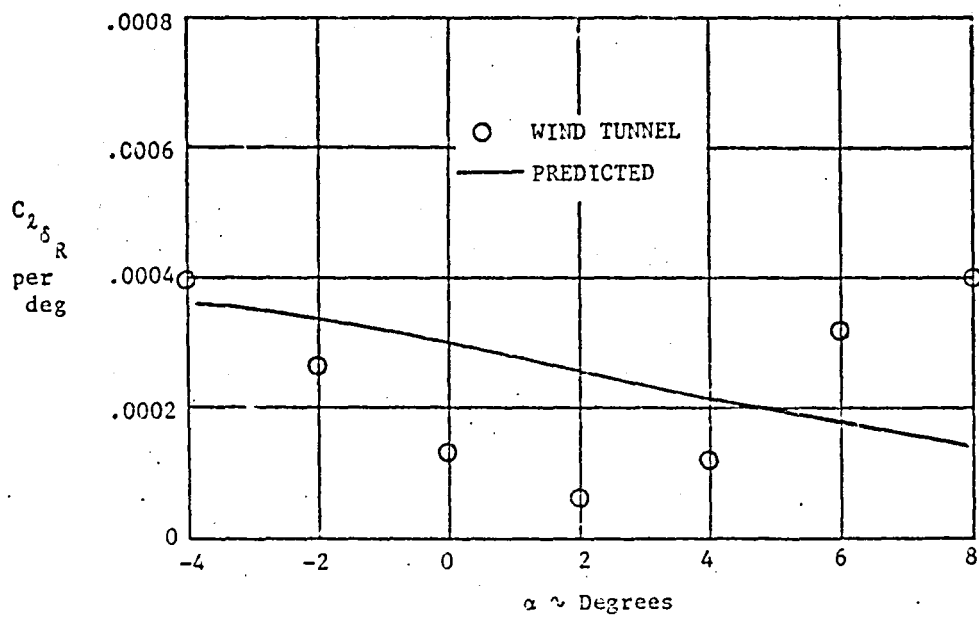


Figure 3.2.5: Comparison of predicted  $C_{2\delta_R}$  with wind tunnel data  
 $(\beta = 0^\circ, \delta_R = \pm 10^\circ)$

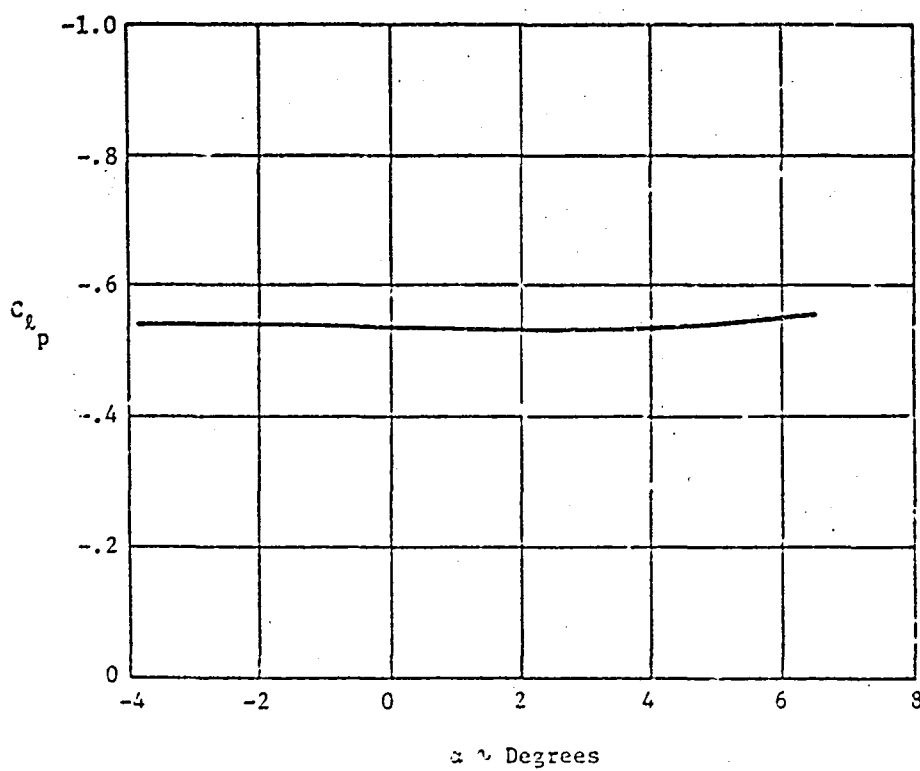


Figure 3.3.1: Predicted  $C_{L_p}$  of the ATLIT Airplane  
(propellers removed)



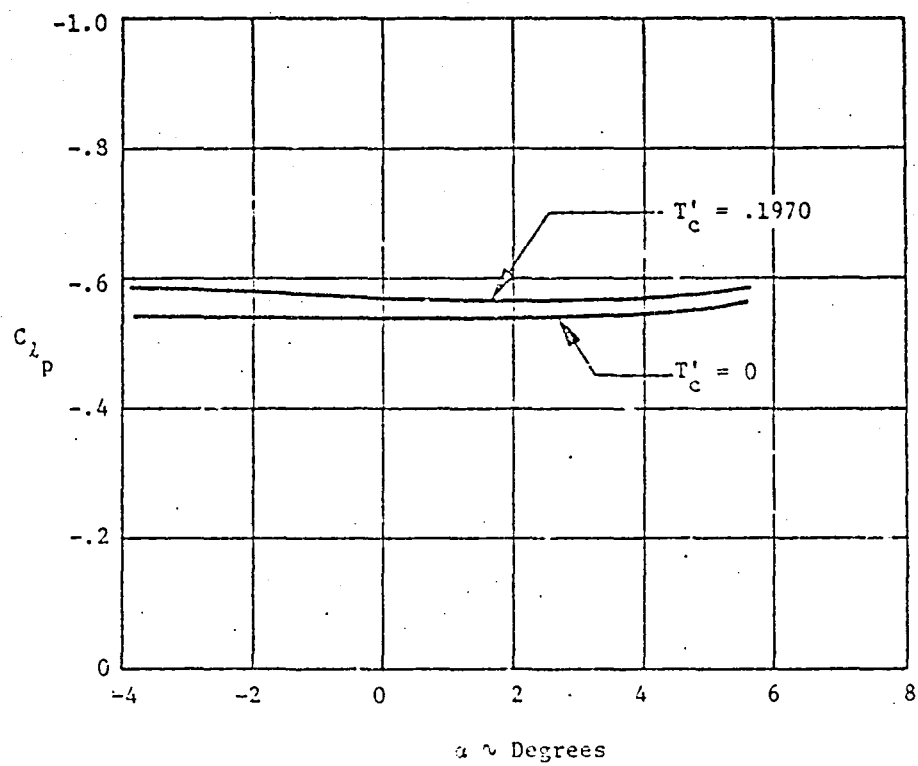


Figure 3.3.2: Effect of power on predicted  $C_{Lp}$  of the ATLIT Airplane

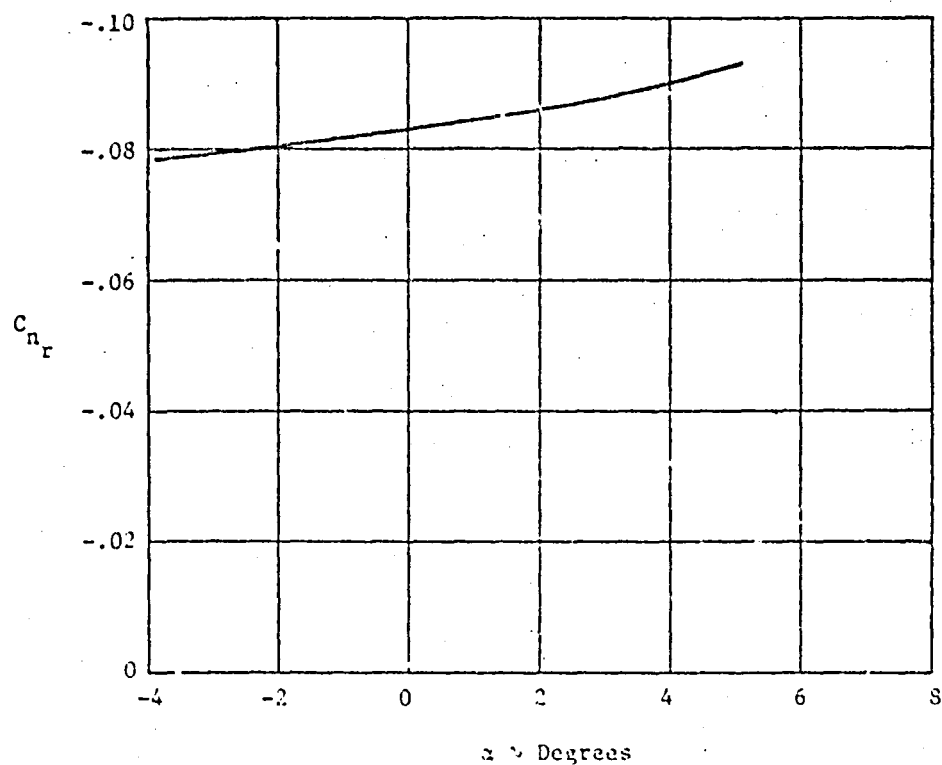


Figure 3.3.3: Predicted  $C_{n_r}$  of the ATLIT Airplane  
(propellers removed)

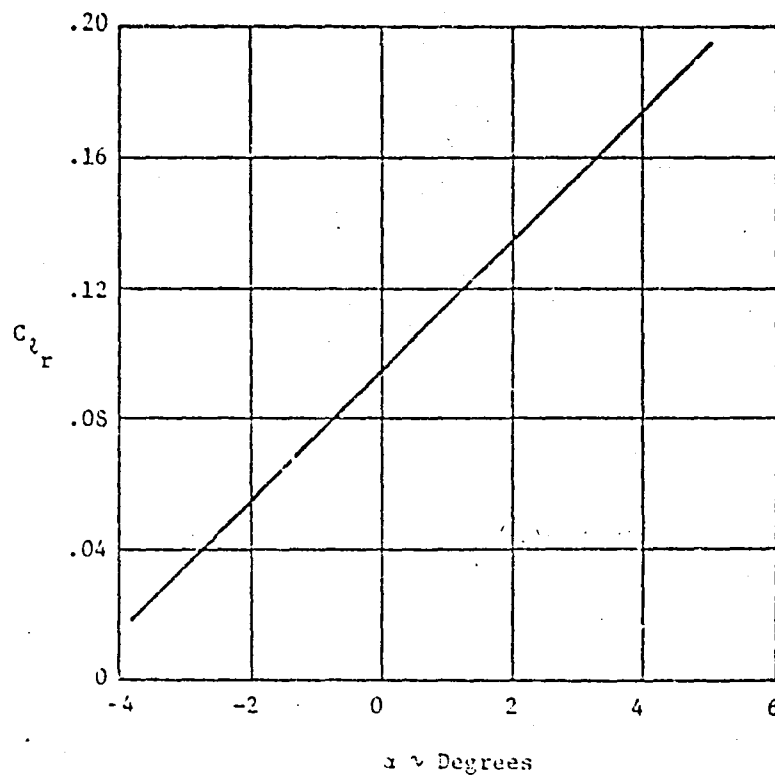


Figure 3.3.4: Predicted  $C_{D_r}$  of the ATLIT Airplane  
(propellers removed)

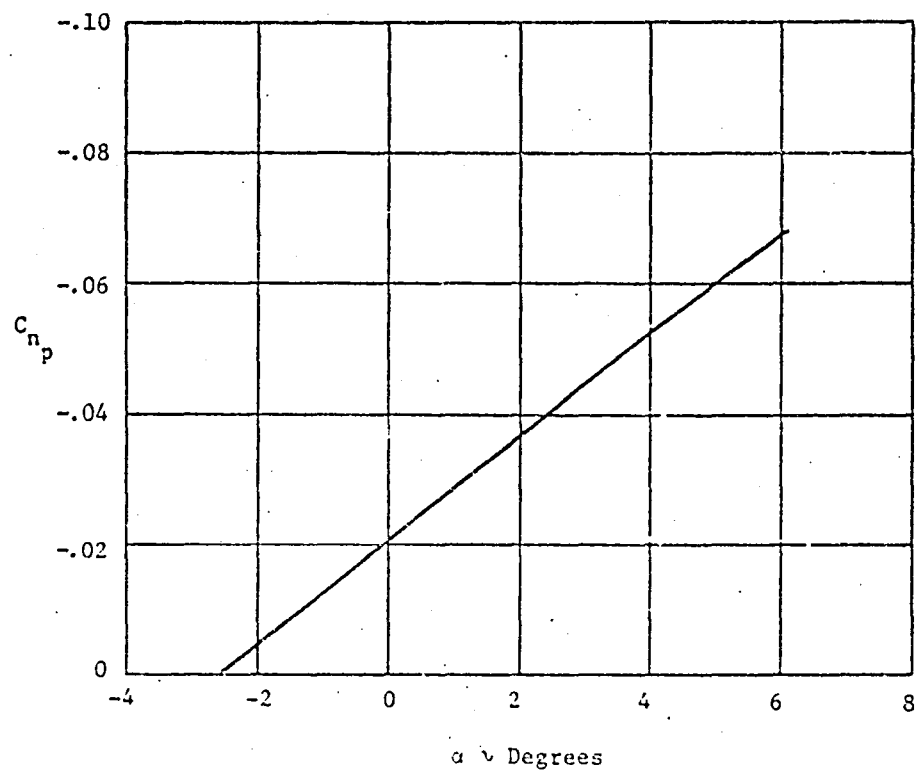


Figure 3.3.5: Predicted  $C_{np}$  of the ATLIT Airplane  
(propellers removed)

## CHAPTER 4

### PREDICTION OF PROPELLER-OFF STATIC STABILITY AND CONTROL CHARACTERISTICS

In this chapter the propeller-off static stability and control characteristics will be discussed. The methods of Reference 3 are used for most all predictions. However, since Reference 3 does not include methods for predicting spoiler control characteristics, Reference 4 will be used.

#### 4.1 Side Force Derivative, $C_{Y\beta}$

The side force due to sideslip derivative,  $C_{Y\beta}$ , of the complete airplane is found by considering the contributions of the following components:

- (1) Wing, including dihedral
- (2) Fuselage, including wing-fuselage interference
- (3) Nacelles
- (4) Vertical tail, including the interference of the wing, fuselage, and horizontal tail.

##### 4.1.1 Wing Contribution, $(C_{Y\beta})_{w_{\Gamma=0}} + (C_{Y\beta})_{\Gamma}$

The wing contribution to  $C_{Y\beta}$  with no dihedral is found from Equation 4.1.1.1 from Reference 3. This expression is valid for low subsonic mach numbers and includes compressibility effects.

$$(C_{Y\beta})_{w_{\Gamma=0}} = \frac{C_{L_w}^2}{57.3} \frac{\delta \tan (\Lambda_{c/4})_w \sin (\Lambda_{c/4})_w}{\pi A_w [\Lambda_w^3 + 4 \cos (\Lambda_{c/4})_w]} \quad (4.1.1.1)$$

where

$C_{L_w}$  is the wing alone lift coefficient obtained from Figure 4.1.1.1

$A_w$  is the aspect ratio of the wing

$\Lambda_{c/4}$  is the quarter-chord sweep of the wing

$$B_2 = \sqrt{1 - M^2 \cos^2 \Lambda_{c/4}}$$

where  $M$  is the Mach Number.

For typical general aviation aircraft with essentially straight wings and moderate aspect ratio ( $A_w > 6$ ), this contribution is insignificant. For the ATLIT airplane

$$(C_{Y_B})_{w_r=0} = 2.32 \times 10^{-7} C_{L_w}^2 \text{ per deg.}$$

The increment in side-force derivative due to wing dihedral may be approximated at low subsonic speeds by the following equation from Reference 3:

$$(C_{Y_B})_r = -0.0001 r \quad (4.1.1.3)$$

For the ATLIT airplane the wing dihedral angle  $r$  is  $7.0^\circ$ . Therefore,

$$(C_{Y_B})_r = -0.0007 \text{ per deg.}$$

The summary calculation of the wing contributions to  $C_{Y_B}$  are given in Table 4.1.1.1.

#### 4.1.2 Fuselage Contribution to $C_{Y_B}$

The fuselage side-force due to sideslip contribution is the result of the side forces produced by the body and the wing-body interference effects. The fuselage alone is the main contributor and is affected by the size and shape of the body. The wing-body interference contribution

is considered to be only a function of wing location. The total fuselage contribution to  $C_{Y_\beta}$  at subsonic Mach numbers is given by Equation 4.1.2.1 from Reference 3 (based on  $S_w$ ).

$$(C_{Y_\beta})_f = K_1 (C_{Y_\beta})_{f_v}^{-2/3} \left( \frac{V}{S_w} \right)^{2/3} \text{ per deg} \quad (4.1.2.1)$$

where

$(C_{Y_\beta})_{f_v}^{-2/3}$  is the contribution of the fuselage only based on two-thirds of the fuselage volume and is equal to but opposite in sign to the potential flow part of the fuselage lift curve slope from Section 4.3 of Reference 1.

$K_1$  is the wing-body interference factor obtained from Figure 4.1.2.1

Table 4.1.2.1 summarizes the effects of the fuselage on  $C_{Y_\beta}$ . For the ATLIT airplane

$$(C_{Y_\beta})_f = -.00299 \text{ per deg.}$$

#### 4.1.3 Nacelle Contribution to $C_{Y_\beta}$

Due to a lack of definitive methods for calculating the contribution of the nacelles to  $C_{Y_\beta}$ , Reference 3 presents the following empirical procedures:

- (1) The effective nacelle length is considered to extend only to the wing leading edge (see Figure 2.1.4).
- (2) The nacelle contribution is approximated from Equation 4.1.3.1 for bodies of circular cross section (based on  $S_w$ ).

$$(C_{Y_\beta})_n = -n_n \frac{2(k_2 - k_1)(S_n)_{\max}}{57.3 S_w} \text{ per deg} \quad (4.1.3.1)$$

where

$n_n$  is the number of nacelles

$S_{x_n}$  is the effective cross section area equal to a circular cross section with a diameter equal to the maximum depth of the nacelle,  $d_n$ , from Figure 2.1.4.

$(k_2 - k_1)$  is the reduced mass factor obtained from Figure 4.1.3.1 as a function of effective length and maximum depth.

(3) To account for flow interference effects, the  $(C_{Y_B})_n$  predicted by Equation 4.1.3.1 is reduced by one-third.

$$(C_{Y_B})_n = -2/3 n_n \left[ \frac{2(k_2 - k_1)(S_{x_n})_{\max}}{57.3 S_w} \right] \text{ per deg} \quad (4.1.3.2)$$

The calculations for the ATLIT airplane are summarized in Table 4.1.3.1. The total nacelle contribution is

$$(C_{Y_B})_n = -.00048 \text{ per deg}$$

#### 4.1.4 Vertical-Tail Contributions to $C_{Y_B}$

The contribution of the vertical tail to  $C_{Y_B}$  is affected by the location of the horizontal tail, the fuselage crossflow at the tail, and the wing-body induced sidewash.

The horizontal tail can serve to increase the loading on the vertical tail by the so-called "end plate effect" when the horizontal tail is at a relatively high or low position. For mid span positions the effect of the horizontal tail is insignificant.

The effect of the fuselage in sideslip is to increase the local cross-flow velocity across the top of the body. This increased velocity causes an increase in vertical tail effectiveness.



As the sidewash due to a yawed wing is small, the primary sidewash contribution is from the body and arises from the body vortex system produced in sideslip. This vortex system produces lateral velocity components which affect the vertical tail.

The method of Reference 3 accounts for the horizontal tail and cross flow factors by computing an effective aspect ratio. The effective aspect ratio of the vertical tail is

$$A_{v\text{eff}} = A_v \left( \frac{A_{v(f)}}{A_v} \right) \left\{ 1 + K_h \left[ \frac{A_{v(fh)}}{A_{v(f)}} - 1 \right] \right\} \quad (4.1.4.1)$$

where

$A_v$  is the actual geometric aspect ratio of the vertical tail

$\frac{A_{v(f)}}{A_v}$  is the ratio of the aspect ratio of the vertical tail in the presence of the body to that of the isolated vertical tail, obtained from Figure 4.1.4.1(a)

$\frac{A_{v(fh)}}{A_{v(f)}}$  is the ratio of the aspect ratio of the vertical tail in the presence of the horizontal tail and the fuselage to the aspect ratio of the vertical tail in the presence of the fuselage only, obtained from Figure 4.1.4.1(b)

$K_h$  is a factor which accounts for the relative size of the vertical and horizontal tails, obtained from Figure 4.1.4.2

Table 4.1.4.1(a) shows the summary calculations to obtain the effective aspect ratio of the ATLIT airplane. It shows an effective aspect ratio of 2.46 compared to a geometric aspect ratio of 1.19.

The lift curve slope of the vertical tail is found using the standard Polhamus equation (4.1.4.2) by using the effective aspect ratio (based on effective vertical tail area,  $S_v$ )

$$(C_{L_{\alpha}})_{v(hf)} = \frac{2 \pi A_{v \text{ eff}}}{2 + \sqrt{\frac{A_{v \text{ eff}}}{k_v^2} [B_1^2 + \tan^2(\Lambda_{c/2})_v] + 4}} \quad (4.1.4.2)$$

where

$$k_v = \frac{(C_{l_{\alpha}})_v}{2\pi} \quad (4.1.4.3)$$

$(C_{l_{\alpha}})_v$  is the section lift curve slope of the vertical tail obtained from Section 4.1 of Reference 1.

$$B_1^2 = 1 - M^2 \quad (4.1.4.4)$$

M is the Mach Number

$(\Lambda_{c/2})_v$  is the mid-chord sweep of the vertical tail.

Table 4.1.4.1(b) shows the summary calculations. For the ATLIT airplane (based on  $S_v$ )

$$(C_{L_{\alpha}})_{v(fh)} = .0494 \text{ per deg}$$

The total vertical tail contribution to  $C_{Y_{\beta}}$  is given in Equation 4.1.4.5 from Reference 3. It modifies the vertical tail lift curve slope to account for wing wake and body sidewash.

$$(C_{Y_{\beta}})_{v(wfh)} = -k_1' (C_{L_{\alpha}})_{v(fh)} \left(1 + \frac{\partial \sigma}{\partial \beta}\right) \frac{\bar{q}_v}{\bar{q}_{\infty}} \frac{S_v}{S_w} \quad (4.1.4.5)$$

where

$k_1'$  is factor which accounts for the relative size of the body near the vertical tail to the size of the tail, from Figure 4.1.4.3.

$$\left(1 + \frac{\partial \sigma}{\partial \beta}\right) \frac{\bar{q}_v}{\bar{q}_{\infty}} = 0.725 + 3.06 \frac{S_v/S_w}{1 + \cos(\Lambda_{c/4})_v} + 0.4 \frac{z_v}{(w_f)_w} + 0.009 A_w \quad (4.1.4.6)$$

where

$(\Lambda_{c/4})_v$  is the quarter-chord sweep of the vertical tail

$z_w$  is the vertical distance from the centerline of the equivalent fuselage to the quarter-chord point of the root chord of the exposed wing panel, obtained from Figure 2.1.4.

$(w_f)_w$  is the depth of the equivalent circular fuselage at the wing, obtained from Figure 2.1.4.

Table 4.1.4.1(c) shows the summary calculations used to find the vertical tail contribution to  $C_{Y_\beta}$ . For the ATLIT airplane (based on  $S_w$ )

$$(C_{Y_\beta})_{v(wfh)} = -0.0056 \text{ per deg}$$

#### 4.1.5 $C_{Y_\beta}$ of the ATLIT Airplane

The total side-force derivative of the complete airplane (propellers off) is

$$C_{Y_\beta} = (C_{Y_\beta})_{w\Gamma=0} + (C_{Y_\beta})_\Gamma + (C_{Y_\beta})_f + (C_{Y_\beta})_n + (C_{Y_\beta})_{v(wfn)} \quad (4.1.5.1)$$

Table 4.1.5.1 summarizes the contribution of each component. For the ATLIT airplane

$$(C_{Y_\beta})_{\text{prop off}} = -0.0098 \text{ per deg}$$

The predicted and experimental  $C_{Y_\beta}$  is shown in Figure 4.1.5.1. In general it shows fairly good agreement, although it is obvious that the effect of angle of attack has not been properly accounted for. These effects are probably in the wing-body interference and sidewash contributions.

Table 4.1.1.1: Wing Contribution to  $C_{Y_3}$

Symbol	Description	Reference	Magnitude
M	Mach number	-	0.081
$A_w$	Wing aspect ratio	Table 2.1.1	10.32
$(\Lambda_{c/4})_w$	Wing quarter-chord sweep angle, deg	Table 2.1.1	1.835
$\Gamma$	Geometric dihedral angle, deg	Table 2.1.1	7
$C_{L_w}$	Wing lift coefficient	Figure 4.1.1.1	variable
Summary: $(C_{Y_3})_{w, \Gamma=0} + (C_{Y_3})_{\Gamma} = 2.32 \times 10^{-7} C_{L_w}^2 - 0.0007$ per deg			

Table 4.1.2.1: Fuselage Contribution to  $C_{Y_3}$

Symbol	Description	Reference	Magnitude
$\frac{2z}{(u_f)_w}$	Wing-body position parameter	Figure 2.1.4	0.326
$K_I$	Wing-body interference factor	Figure 4.1.2.1	1.16
$(C_{Y_3})_{f_v}^{2/3}$	$C_{Y_3}$ of equivalent circular fuselage based on 2/3 power of fuselage volume [considered equal to the negative of the potential flow part of $(C_{L_a})_{f_v}^{2/3}$ ] times the fuselage volume to 2/3 power	Table 4.3.1 Reference 1	0.002574
$S_w$	Wing reference area, $m^2$ ( $ft^2$ )	Table 2.1.1	14.40 (155.0)
Summary: $(C_{Y_3})_f = -0.002986$ per deg			

Table 4.1.3.1: Nacelle Contribution to  $C_{Y_3}$

Symbol	Description	Reference	Magnitude
$n$	Number of nacelles	-	2
$S_w$	Wing reference area, $m^2$ ( $ft^2$ )	Table 2.1.1	14.40 (155.0)
$(S_{x_n})_{max}$	Effective nacelle cross-section area, $m^2$ ( $ft^2$ )	Figure 2.1.4	0.292 (3.14)
$\frac{l_n}{d_n}$	Effective nacelle fineness ratio	Figure 2.1.4	2.08
$(k_2 - k_1)$	Nacelle reduced mass factor	Figure 4.1.3.1	0.504
Summary: $(C_{Y_3})_n = -0.000476$ per deg			

4.1.4.1: Vertical Tail Contribution to  $C_{Y\delta}$   
(a) Effective Aspect Ratio

Symbol	Description	Reference	Magnitude
$S_h$	Horizontal tail area, $m^2$ ( $ft^2$ )	Table 2.1	3.60 (38.7)
$S_v$	Vertical tail area, $m^2$ ( $ft^2$ )	Table 2.1	1.85 (19.9)
$b_v$	Vertical tail span, $m$ ( $ft$ )	Table 2.1	1.52 (5.0)
$A_v$	Vertical tail aspect ratio	Table 2.1	1.19
$(c_v)_h$	Vertical tail chord at horizontal tail, $m$ ( $ft$ )	Figure 2.1.5	1.80 (5.92)
$x_{ac_h}(c_v)_e$	Distance from leading edge of vertical tail to a.c. of horizontal tail, in plane of horizontal tail, $m$ ( $ft$ )	Figure 2.1.5	1.45 (4.75)
$z_{c_{r_h}}(c_{r_v})$	Distance from root chord of vertical tail to root chord of horizontal tail, $m$ ( $ft$ )	Figure 2.1.5	-0.051 (-0.167)
$(d_f)_v$	Depth of fuselage at quarter-root chord of vertical tail, $m$ ( $ft$ )	Figure 2.1.5	0.661 (2.17)
$\frac{A_v(f)}{A_v}$	Ratio of vertical tail aspect ratio in presence of fuselage to isolated vertical tail aspect ratio	Figure 4.1.4.1 (a)	1.5
$\frac{A_v(fh)}{A_v(f)}$	Ratio of vertical tail aspect ratio in presence of fuselage and horizontal tail to aspect ratio of tail in presence of fuselage alone	Figure 4.1.4.1 (b)	1.25
$K_h$	Relative tail size factor	Figure 4.1.4.2	1.12
Summary: $A_{v_{eff}} = 2.46$			

Table 4.1.4.1: (continued)  
(h) Vertical Tail Lift Curve Slope

Symbol	Description	Reference	Magnitude
$M$	Mach number	-	0.081
$S_1^2$	$1 - M^2$	Equation 4.1.4.4	0.991
$A_{v,eff}$	Effective vertical tail aspect ratio	Table 4.1.4.1 (a)	2.46
$(\Lambda_{c/2})_v$	Vertical tail half-chord sweepback, deg	Table 2.1.1	29.3
$(C_{l_\alpha})_v$	Vertical tail section lift curve slope, per rad	Table 4.1.2 Reference 1	6.246
$k_v$	$\frac{(C_{l_\alpha})_v}{S_v}$	Equation 4.1.4.3	0.988
Summary: $(C_{L_\alpha})_{v(th)} = 0.0444$ per deg = 2.83 per rad (based on $S_v$ )			



Table 4.1.4.1: Concluded  
(c) Vertical Tail Contribution to  $C_{Y_3}$

Symbol	Description	Reference	Magnitude
$S_w$	Wing reference-area, $m^2$ ( $ft^2$ )	Table 2.1	14.40 (155.0)
$(\Lambda_{c/4})_v$	Vertical tail quarter-chord sweep angle, deg	Table 2.1.1	34.5
$z_w$	Distance from equivalent fuselage centerline	Figure 2.1.4	0.199 (0.638)
$(w_F)_w$	Width of equivalent fuselage at wing, $m$ ( $ft$ )	Figure 2.1.4	1.19 (3.92)
$A_w$	Wing aspect ratio	Table 2.1	10.32
$(1 + \frac{3z_w}{S_w}) \frac{d_w}{A_w}$	Wing wake and fuselage side wash factor	Equation 4.1.4.6	1.097
$k_1^*$	Relative body size to tail size parameter	Figure 4.1.4.3	0.80
Summary: $(C_{Y_3}^*)_{v(tch)} = -0.0056$ per deg = $-0.319$ per rad (based on $S_w$ )			

Table 4.1.5.1:  $C_{Y_B}$  of the ATLIT Airplane

Symbol	Description	Reference	Magnitude
$(C_{Y_B})_{\alpha=0}$	Wing contribution without dihedral, per deg	Table 4.1.1.1	$2.32 \times 10^{-7} C_{L_W}^2$
$(C_{Y_B})_i$	Increment due to dihedral, per deg	Table 4.1.1.1	-0.0007
$(C_{Y_B})_f$	Fuselage contribution, per deg	Table 4.1.2.1	-0.00299
$(C_{Y_B})_n$	Nacelle contribution, per deg	Table 4.1.3.1	-0.000476
$(C_{Y_B})_{v(vfn)}$	Vertical tail contribution with interference accounted for, per deg	Table 4.1.4.1(c)	-0.0056
Summary: $(C_{Y_B})_{prop off} = -0.00977$ per deg			

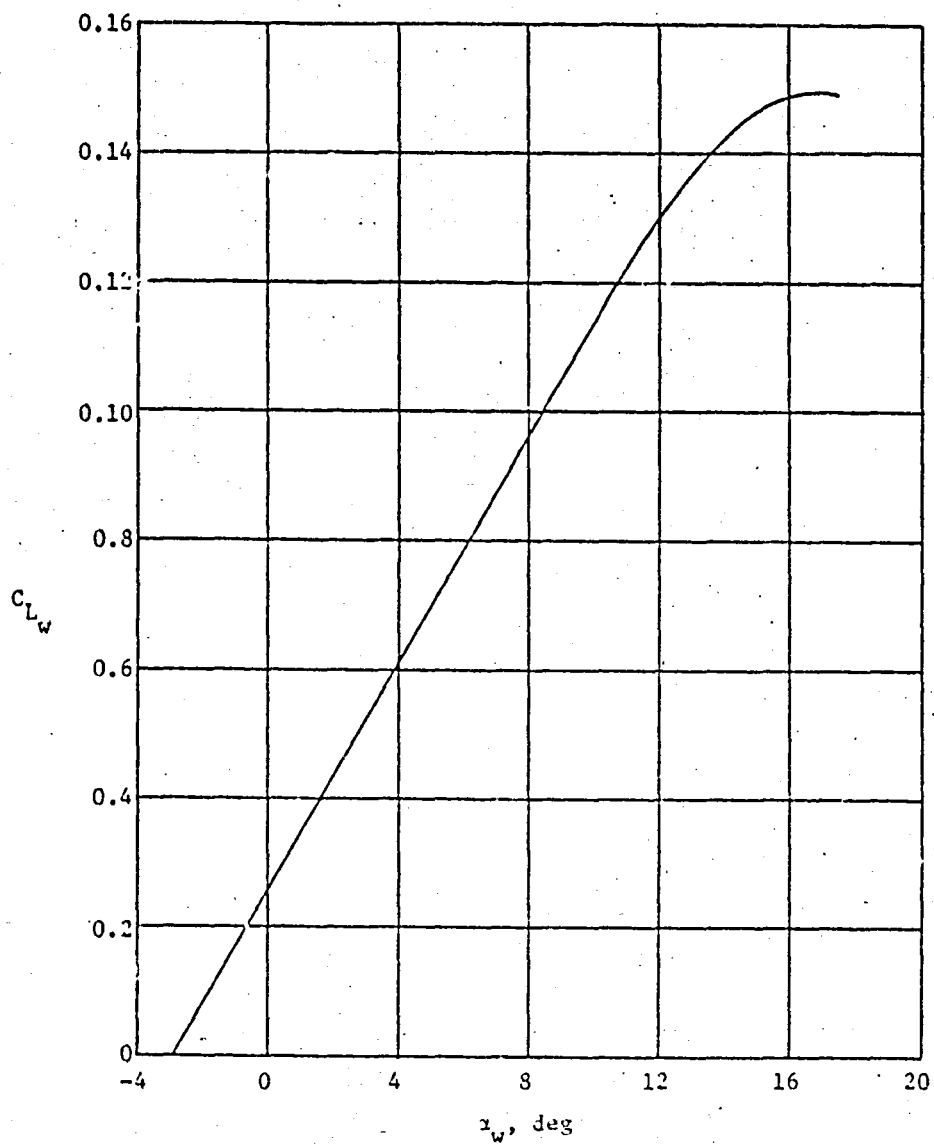


Figure 4.1.1.1: Propeller off lift curve of the wing alone for the ATLIT airplane (Reference 1)

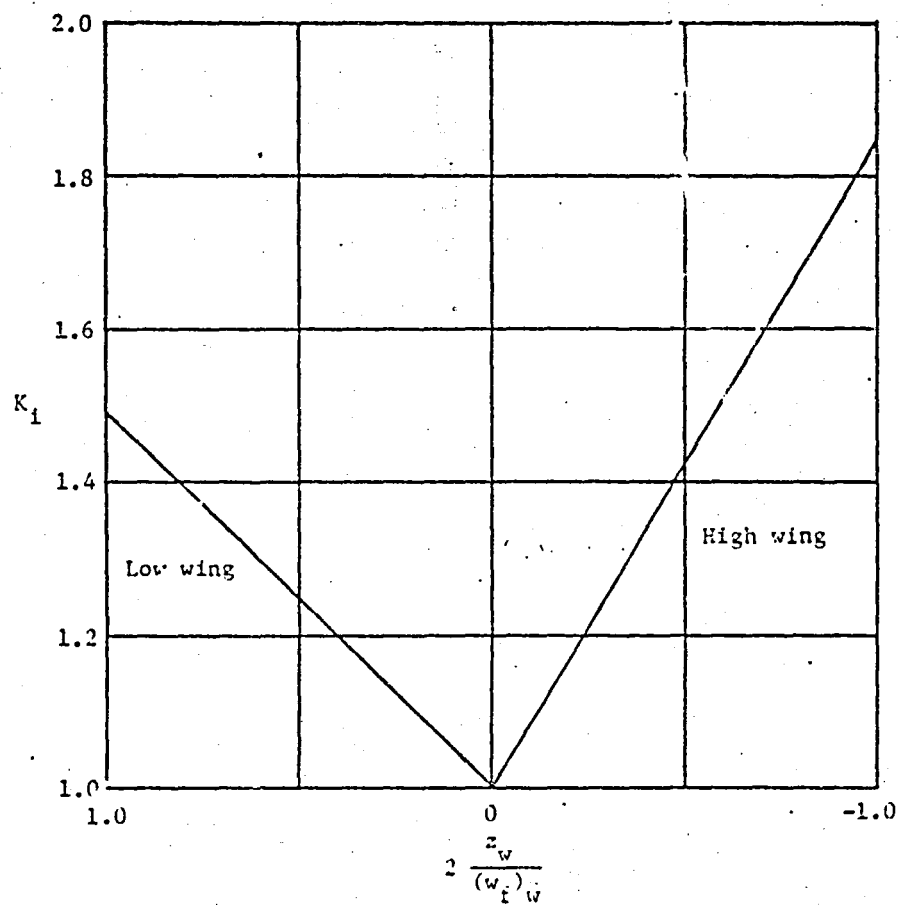


Figure 4.1.2.1: Wing body interference factor (Reference-3)

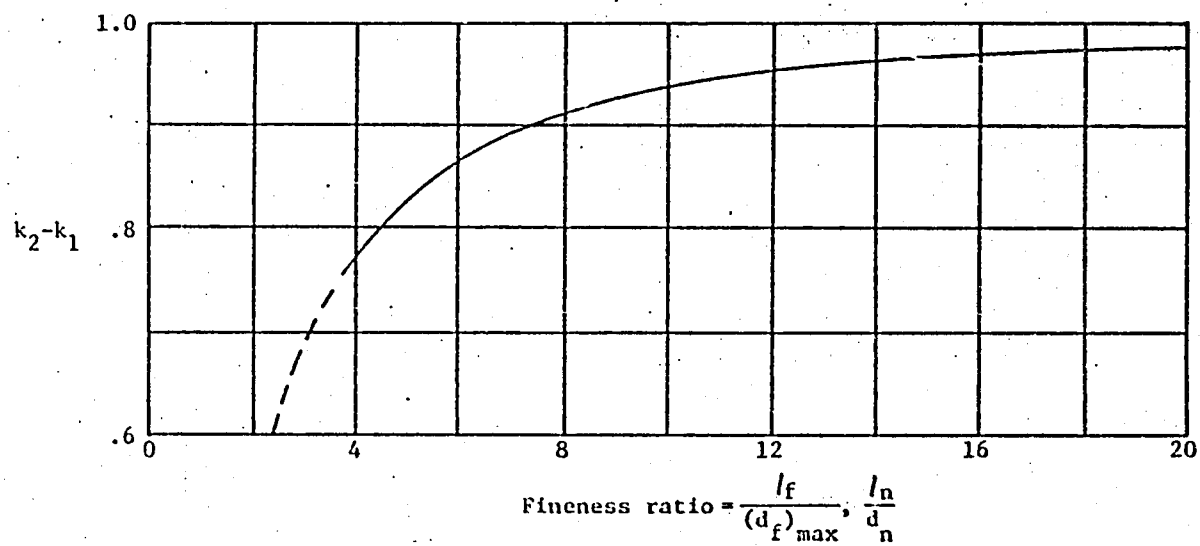


Figure 4.1.3.1: Nacelle reduced mass factor (Reference 3)

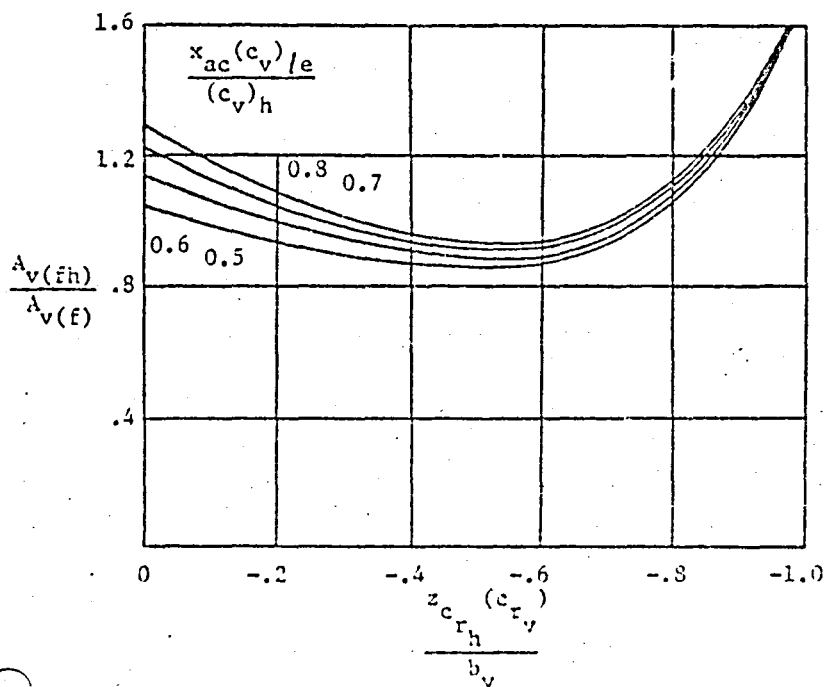
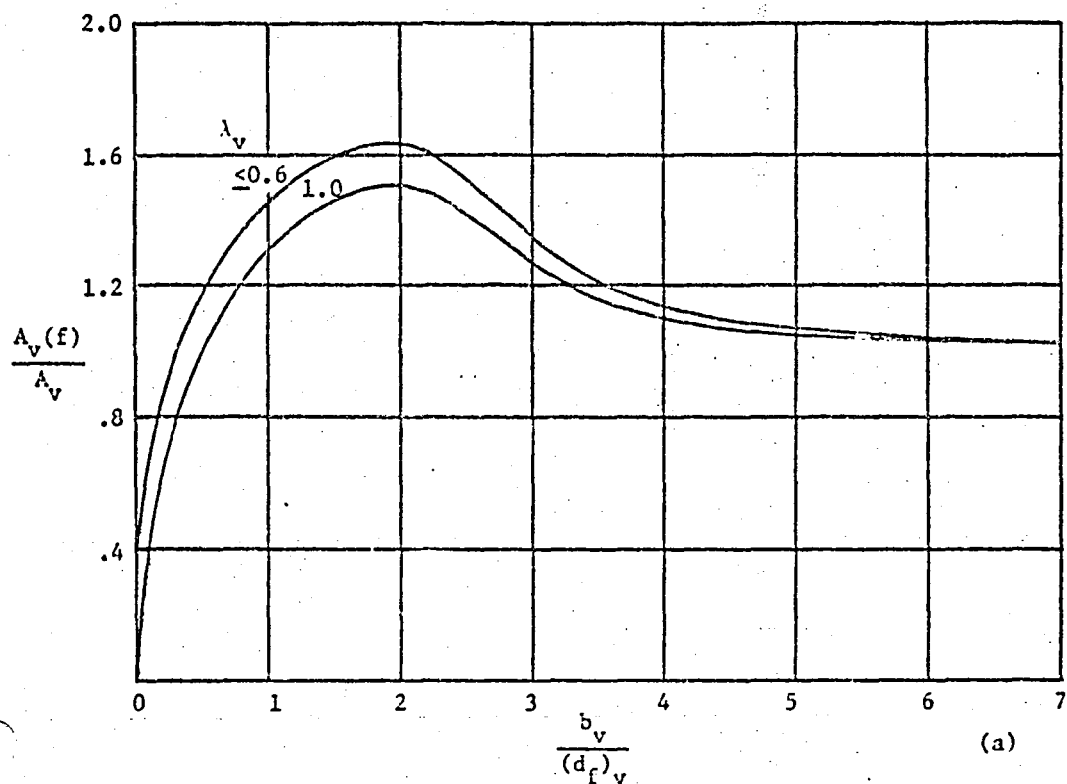


Figure 4.1.4.1 (a) & (b):

Vertical tail aspect ratio  
interference factors  
(Reference 3)

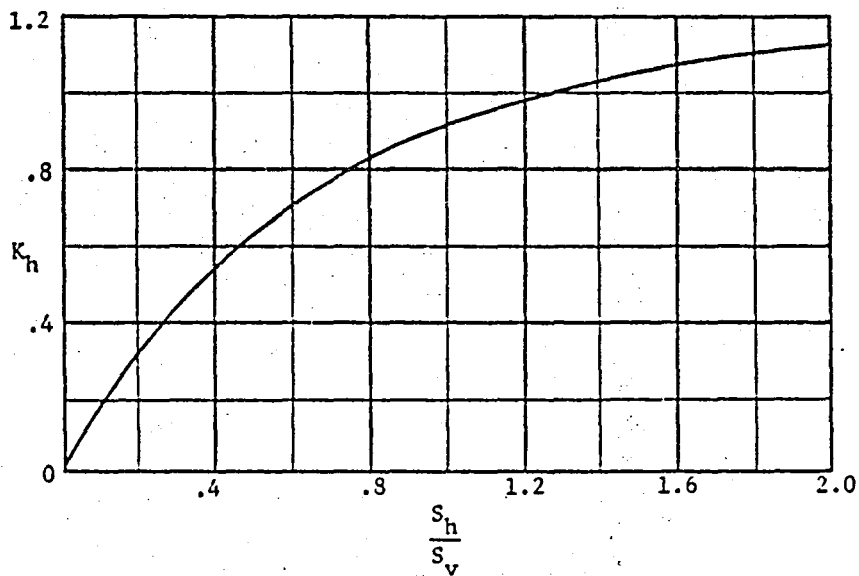


Figure 4.1.4.2: Relative tail size factor (Reference 3)

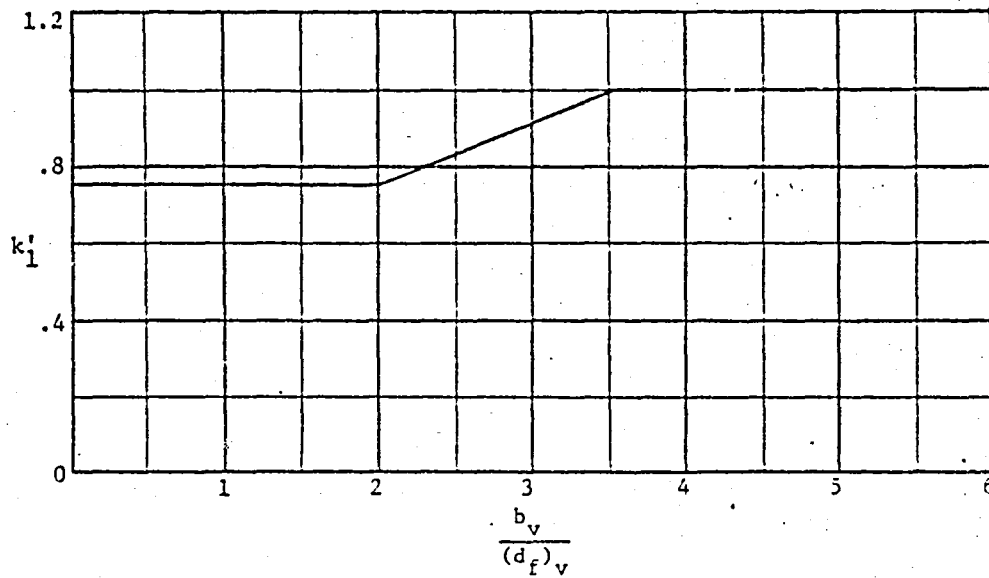


Figure 4.1.4.3: Relative body size factor (Reference 3)

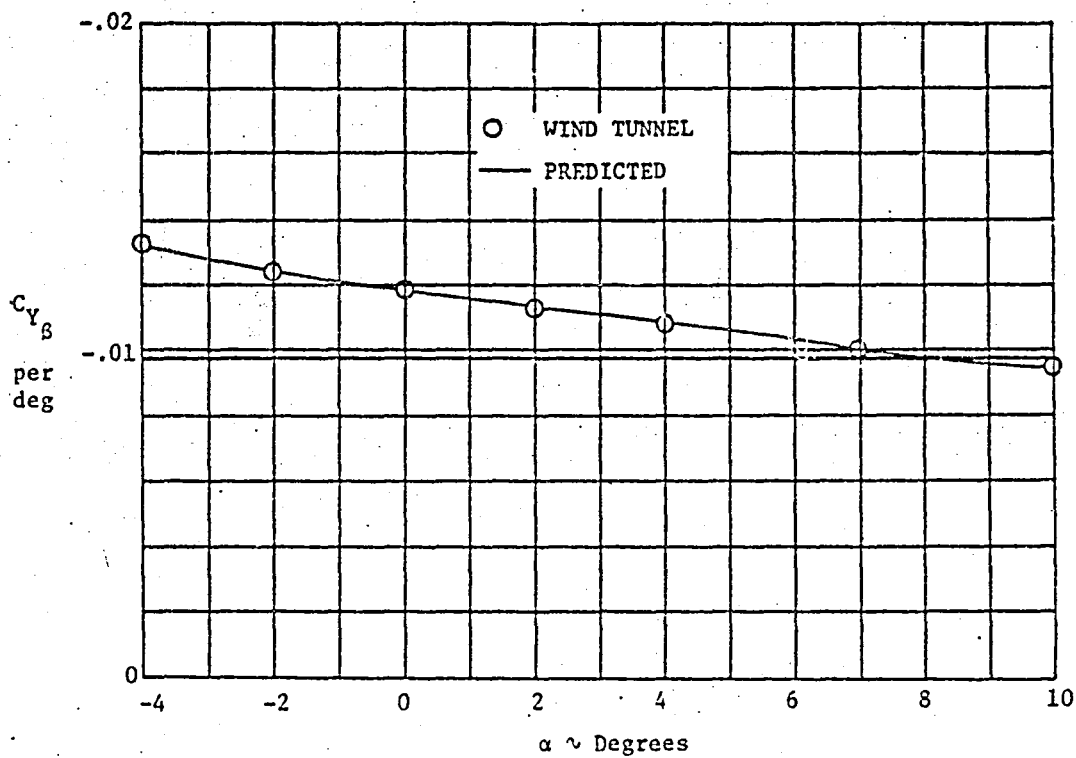


Figure 4.1.5.1: Comparison of predicted  $C_{Y\beta}$  with full scale wind tunnel data (propellers removed,  $N_{Re} = 2.3$  million).



#### 4.2 Yawing Moment Derivative, $C_{n\beta}$

The yawing moment due to sideslip derivative,  $C_{n\beta}$ , of the complete airplane is found by considering the contributions of the following components:

- (1) Wing
- (2) Fuselage, including wing-fuselage interference
- (3) Nacelles
- (4) Vertical tail, including the interference and sidewash of the vertical tail.

##### 4.2.1 Wing Contribution to $C_{n\beta}$

The wing contribution to  $C_{n\beta}$  is due to the increased induced drag on the leading wing caused by increased lift on the leading wing.

Equation 4.2.1.1 from Reference 3 gives the incompressible  $(C_{n\beta})_w$

$$\frac{C_{n\beta}}{C_L^2} = \frac{1}{57.3} \left[ \frac{1}{4\pi A_w} - \frac{\tan \Lambda_{c/4}}{\pi A_w (A_w + 4 \cos \Lambda_{c/4})} \left( \cos \Lambda_{c/4} - \frac{A_w^2}{2} - \frac{A_w^2}{3 \cos \Lambda_{c/4}} + 6 \frac{\bar{x}}{\bar{c}_w} \frac{\sin \Lambda_{c/4}}{A_w} \right) \right] \quad (4.2.1.1)$$

where

$A_w$  is the wing aspect ratio

$\Lambda_{c/4}$  is the quarter chord sweep angle of the wing

$\bar{c}_w$  is the wing mean aerodynamic chord

$\bar{x}$  is the distance (positive rearward) from the center of gravity to the wing aerodynamic center along the mean chord line.

To correct for the three-dimensional effects of compressibility, Equation 4.2.1.2 from Reference 3 gives the wing contribution to  $C_{n_3}$ .

$$\left(\frac{C_{n_3}}{C_L^2}\right)_H = \left(\frac{A_w + 4 \cos \Lambda_{c/4}}{A_w B_2 + 4 \cos \Lambda_{c/4}}\right) \left(\frac{A_w^2 B_2^2 + 4 A_w B_2 \cos \Lambda_{c/4} - 8 \cos^2 \Lambda_{c/4}}{A_w^2 + 4 A_w \cos \Lambda_{c/4} - 8 \cos^2 \Lambda_{c/4}}\right) \left(\frac{C_{n_3}}{C_L^2}\right)_{\text{low speed}} \quad (4.2.1.2)$$

where

$$B_2 = \sqrt{1 - M^2 \cos^2 \Lambda_{c/4}}$$

$C_{L_w}$  is the wing lift coefficient obtained from Figure 4.1.1.1

The summary calculations of the wing contributions to  $C_{n_3}$  are shown in Table 4.2.1.1. For the ATLIT airplane

$$(C_{n_3})_w = 0.000156 C_{L_w}^2 \text{ per deg} \quad (4.2.1.3)$$

#### 4.2.2 Fuselage Contribution to $C_{n_3}$

The fuselage plus wing-body interference contribution to  $C_{n_3}$  is given in Equation 4.2.2.1, taken from Reference 3. In general, wing sweep, wing taper, and Mach number have no effect.

$$(C_{n_3})_{f(w)} = -K_N \frac{(S_f)_s}{S_w} \frac{l_f}{b_w} \quad (4.2.2.1)$$

where

$(S_f)_s$  is the fuselage side area obtained from Figure 2.1.4

$S_w$  is the wing area

$b_w$  is the wing span

$l_f$  is the length of the fuselage from Figure 2.1.4.

There are two ways to obtain the correlation factor  $K_N$ . In Reference 4,  $K_N$  is given for midwing configurations and it is assumed that  $(C_{n_3})_{f(w)}$  does not vary with angle of attack. A nomograph for obtaining  $K_N$  midwing configurations is shown in Figure 4.2.2.1.

In Reference 3 it was concluded on the basis of wind-tunnel data that angle of attack variations were significant for non midwing configurations. Figure 4.2.2.2 taken from Reference 3 is used to extend the nomograph to obtain  $K_N$  based on angle of attack. This graph was constructed for an airplane with  $2z_w/(w_f)_w = 0.50$  and is based on tail-off wind-tunnel data. For the ATLIT airplane  $2z_w/(w_f)_w = 0.33$ ; nevertheless, Figure 4.2.2.2 was used to obtain  $K_N$  in the absence of tail-off wind-tunnel data. If tunnel data is available, the procedure outlined in Reference 3 could be used to derive  $K_N$  for other wing locations.

Table 4.2.2.1 shows the summary calculation for the fuselage contribution to  $C_{n_3}$ .

#### 4.2.3 Nacelle Contribution to $C_{n_3}$

The nacelles contribution to  $C_{n_3}$  derives directly from the nacelle side force acting through the nacelles moment arm. Equation 4.2.3.1 from Reference 3 is used to obtain the nacelles contribution.

$$(C_{n_3})_n = (C_{Y_3})_n \left( \frac{x_n \cos \alpha + z_n \sin \alpha}{b_w} \right) \quad (4.2.3.1)$$

where

$(C_{Y_3})_n$  is the nacelles contribution to side force due to side slip, from Section 4.1.3.

$x_n$  is the longitudinal distance from the nacelle center of pressure to the airplane center of gravity, obtained from Figure 2.1.4

$z_n$  is the vertical distance from the nacelle center of pressure to the airplane center of gravity, from Figure 2.1.4

$b_w$  is the wing span, obtained from Table 2.1.

Table 4.2.3.1 summarizes the contribution of the nacelles to  $C_{n_3}$ .

#### 4.2.4 Vertical-Tail Contribution to $C_{n_3}$

The vertical tail contribution to  $C_{n_3}$  is obtained in a similar manner to the nacelle contribution. The vertical tail  $C_{n_2}$  is the vertical tail  $C_{Y_3}$  times the vertical tail moment arm and is obtained from Equation 4.2.4.1 from Reference 3.

$$(C_{n_3})_{v(wfh)} = -(C_{Y_3})_{v(wfn)} \left( \frac{l_v \cos \alpha - z_v \sin \alpha}{b_w} \right) \quad (4.2.4.1)$$

where

$(C_{Y_3})_{v(wfn)}$  is the side force due to sideslip of the vertical tail, from Section 4.1.4

$l_v$  is the distance from the center of gravity to the quarter chord of the mean aerodynamic chord of the vertical tail, parallel to the X-body axis, obtained from Figure 2.1.3.

$z_v$  is the distance from the center of gravity to the quarter chord of the mean aerodynamic chord of the vertical tail, parallel to the Z-body axis, obtained from Figure 2.1.3.

Table 4.2.4.1 summarizes the vertical tail contribution to  $C_{n_3}$ .

#### 4.2.5 $C_{n\beta}$ of the ATLIT Airplane

The total yawing moment derivative of the complete airplane (propellers off) is

$$(C_{n\beta})_{\text{prop off}} = (C_{n\beta})_w + (C_{n\beta})_{f(w)} + (C_{n\beta})_n + (C_{n\beta})_{v(wfh)} \quad (4.2.5.1)$$

Table 4.2.5.1 summarizes the contribution of each component and the total  $C_{n\beta}$ .

Table 4.2.1.1: Wing Contribution to  $C_{n_3}$

Symbol	Description	Reference	Magnitude
$M$	Mach number	-	0.381
$A_w$	Wing aspect ratio	Table 2.1.1	10.92
$\Lambda_{c/4}$	Wing quarter-chord sweep angle, deg	Table 2.1.1	1.315
$c_w$	Mean aerodynamic chord of wing, m (in)	Table 2.1.1	1.225 (48.22)
$\bar{x}$	Distance from wing a.c. to airplane center of gravity; in terms of $\bar{c}_w$	Figure 2.1.1	0
$\beta_2$	$\sqrt{1 - M^2 \cos^2 \Lambda_{c/4}}$	-	.997
$C_{L_w}$	Wing lift coefficient	Figure 4.1.1	variable
Summary: $(C_{n_3})_w = 0.000156 C_{L_w}^2$ per deg.			

Table 4.2.2.1: Fuselage Contribution to  $C_{n3}$

Symbol	Description	Reference	Magnitude
$(S_f)_s$	Fuselage side area, $m^2$ ( $ft^2$ )	Table 4.0.1 Reference 1	7.72 (83.1)
$S_w$	Wing reference area, $m^2$ ( $ft^2$ )	Table 2.1.1	14.40 (155.0)
$l_f$	Fuselage length, $m$ ( $ft$ )	Figure 2.1.4	3.352 (27.40)
$b_w$	Wing span, $m$ ( $ft$ )	Table 2.1.1	12.19 (40.0)
$z_w$	Vertical distance of wing root chord below the centerline of the equivalent fuselage, $m$ ( $ft$ )	Figure 2.1.4	0.194 (0.638)
$(w_f)_w$	Width of the fuselage at the quarter chord of the exposed root chord, $m$ ( $ft$ )	Figure 2.1.4	1.19 (3.92)
$N_{RE}$	Reynolds number based on fuselage length	-	$1.57 \times 10^7$
$x_2, h_1, h_2$	Geometric fuselage parameters	Figure 4.2.2.1	as listed
$K_f$	Correction factor for fuselage contribution	Figure 4.2.2.2	variable

$\alpha$ , Deg	$K_f$ Fig. 4.2.2.2	$(C_{n3})_f$ Eq. 4.2.2.1
-4	0.0015	-0.00011
-2	0.0019	-0.00014
0	0.0024	-0.00018
2	0.0027	-0.00017
4	0.0030	-0.00022
8	0.0018	-0.00051

Table 4.2.3.1: Nacelle Contribution to  $C_{n_3}$

Symbol	Description	Reference	Magnitude
$(C_{n_3})_n$	Nacelle contribution to $C_{n_3}$	Table 4.1.5.1	-0.000476
$x_p$	Distance from c.g. to nacelle center of pressure, m (ft)	Figure 2.1.4	0.966 (3.17)
$z_n$	Distance from X-body axis to nacelle center of pressure, m (ft)	Figure 2.1.4	-0.152 (-.50)
$b_w$	Wing span, m (ft)	Table 2.1.1	12.19 (40.0)
Summary: $(C_{n_3})_n = -0.0000377 \cos \alpha + 0.0000060 \sin \alpha$			



Table 4.2.4.1: Vertical Tail Contribution to  $C_{n_3}$

Symbol	Description	Reference	Magnitude
$(C_{Y_3})_{v(wfn)}$	Vertical tail contribution to $C_{Y_3}$	Table 4.1.5.1	-0.0056
$l_v$	Distance along X-body axis from c.g. to quarter-chord of vertical tail mean aerodynamic chord, m (ft)	Figure 2.1.5	4.50 (14.75)
$z_v$	Distance perpendicular to X-body axis from c.g. to quarter-chord of vertical tail mean aerodynamic chord, m (ft)	Figure 2.1.5	1.22 (4.0)
$b_v$	Wing span, m (ft)	Table 2.1.1	12.19 (40.0)
Summary: $(C_{n_3})_{v(wfn)} = 0.0021 \cos \alpha + 0.0056 \sin \alpha$			

Table 4.2.5.1:  $C_{n_3}$  of the ATLIT Airplane

$\alpha$ , deg	$(C_{n_3})_w$ Table 4.2.1.1	$(C_{n_3})_f$ Table 4.2.2.1	$(C_{n_3})_n$ Table 4.2.3.1	$(C_{n_3})_{v(wfn)}$ Table 4.2.4.1	$(C_{n_3})_{prop\ off}$ Eq. 4.2.5.1
-4	0	-0.00013	-0.00004	0.00206	0.00199
-2	0	-0.00014	-0.00004	0.00208	0.00190
0	0.00001	-0.00016	-0.00004	0.00210	0.00191
2	0.00004	-0.00017	-0.00004	0.00212	0.00195
4	0.00007	-0.00022	-0.00004	0.00213	0.00194

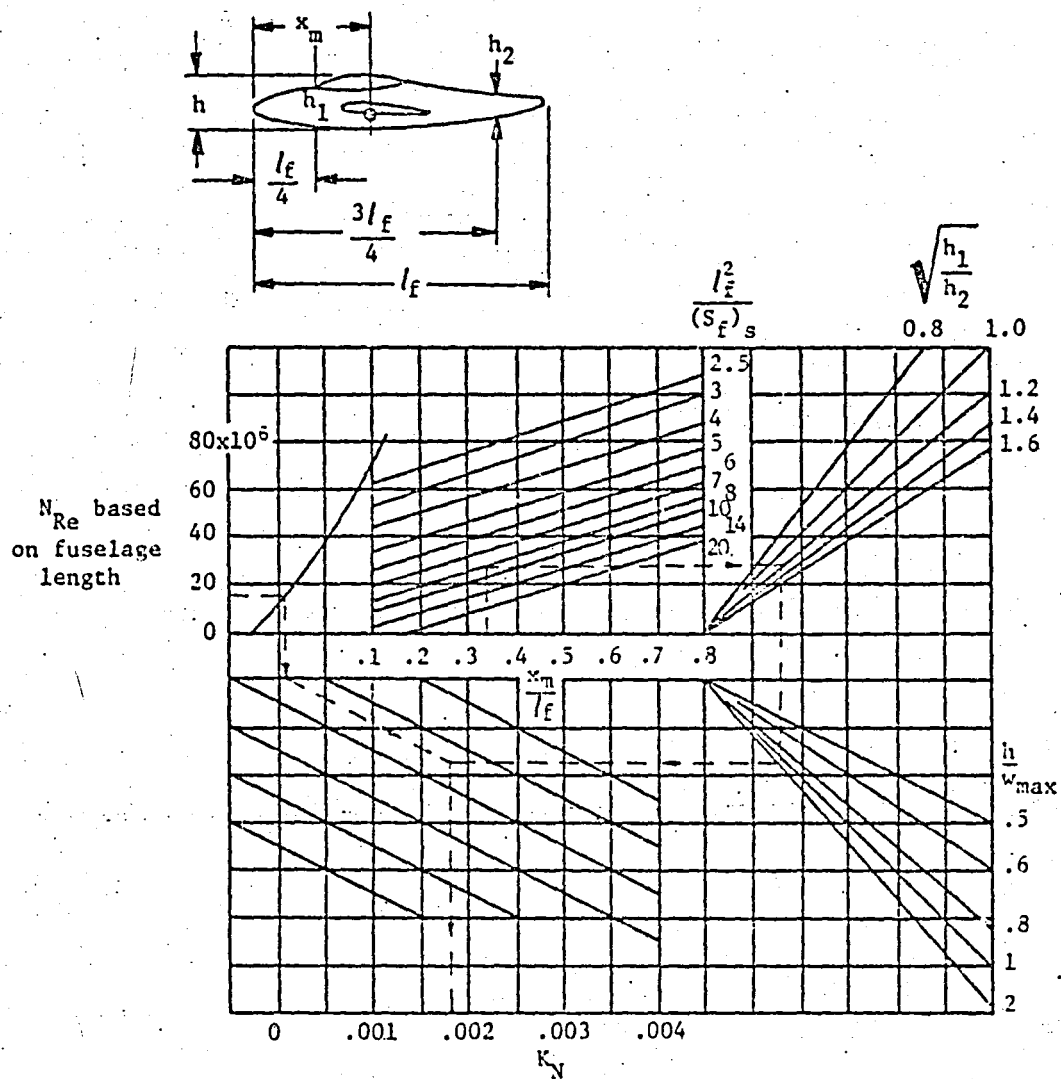
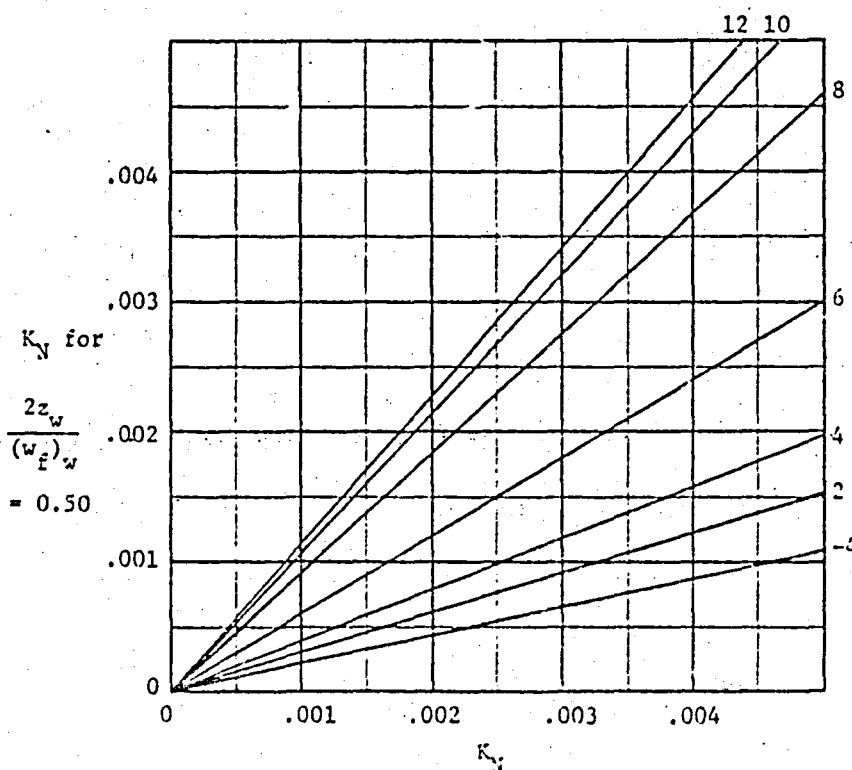
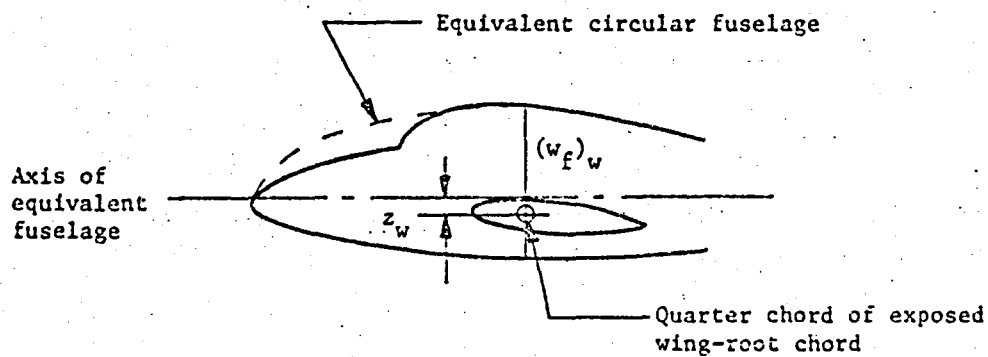


Figure 4.2.2.1: Nomograph for  $K_N$  factor for midwing configurations (Reference 3)



(midwing, Figure 4.2.2.1)

Figure 4.2.2.2: Variation of  $K_Y$  with angle of attack  
(Reference 3)

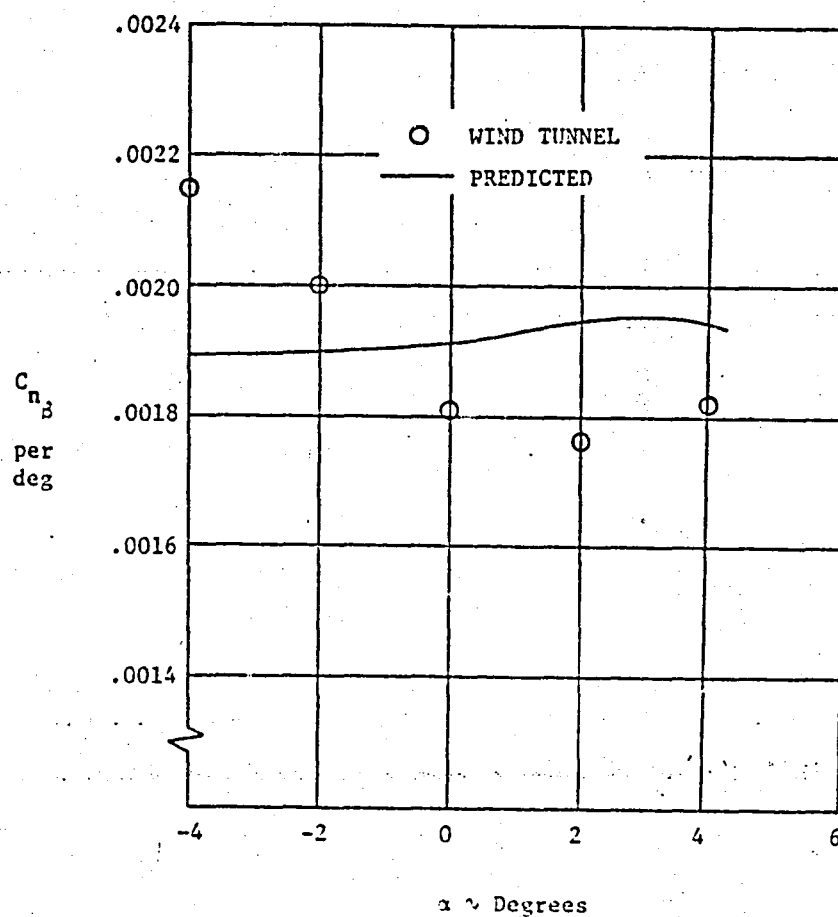


Figure 4.2.5.1: Comparison of predicted  $C_{n_3}$  with full scale wind tunnel data (propellers removed,  $N_{Re} = 2.3$  million)

#### 4.3 Rolling Moment Derivative, $C_{l_\beta}$

The airplane rolling moment due to sideslip,  $C_{l_\beta}$ , is composed of the following contributions:

- (1) Wing (without dihedral plus the effect of dihedral)
- (2) The effects of the fuselage on wing contributions
- (3) The vertical tail contribution.

For general aviation airplanes without large dihedral angle on the horizontal tail, it can be assumed that there is no significant contribution by the horizontal tail.

##### 4.3.1 Wing Contribution to $C_{l_\beta}$

From Reference 3 for subsonic speed in the linear lift range, the wing contribution is given in Equation 4.3.1.1.

$$(C_{l_\beta})_w = (C_{l_\beta})_{w_{\Gamma=0}} + (C_{l_\beta})_{\Gamma} + (C_{l_\beta})_{\theta} \quad (4.3.1.1)$$

Generally the effect of twist  $(C_{l_\beta})_{\theta}$  is not significant.

The wing contribution to  $C_{l_\beta}$  in the absence of geometric dihedral is given by Equation 4.3.1.2 from Reference 3.

$$(C_{l_\beta})_{w_{\Gamma=0}} = C_{L_w} \left[ \left( \frac{C_{l_\beta}}{C_{L_w}} \right)_{M=0} + \left( \frac{\Delta C_{l_\beta}}{C_{L_w}} \right)_M \right] \quad (4.3.1.2)$$

where

$C_{L_w}$  is the wing lift coefficient from Figure 4.1.1.1

$\left( \frac{C_{l_\beta}}{C_{L_w}} \right)_{M=0}$  is the low speed portion obtained from Figure 4.3.1.1.

$\left( \frac{\Delta C_{l_\beta}}{C_{L_w}} \right)_M$  is the correction for compressible flow obtained from Equation 4.3.1.3.

$$\left( \frac{\Delta C_{L_B}}{C_{L_M}} \right)_M = -\frac{1}{2} \bar{y}^* \frac{A_U^2 M^2 \tan \Lambda_{c/4}}{\left[ \left( \frac{A_U}{\cos \Lambda_{c/4}} \right)^2 - A_U^2 M^2 + 4 \right]^{1/2} \left\{ 2 + \left[ \left( \frac{A_U}{\cos \Lambda_{c/4}} \right)^2 - A_U^2 M^2 + 4 \right]^{1/2} \right\}} \quad (4.3.1.3)$$

where  $\bar{y}^*$  is the spanwise location of the centroid of the angle of attack load, as a function of semispan, obtained from Figure 4.3.1.2.

The effect of uniform dihedral is accounted for by Equation 4.3.1.4 from Reference 3.

$$(C_{L_3})_\Gamma = \Gamma \left( \frac{C_{L_B}}{\Gamma} \right)_{M=0} K_{M_\Gamma} \text{ per deg} \quad (4.3.1.4)$$

where

$\Gamma$  is the dihedral angle in degrees

$\left( \frac{C_{L_B}}{\Gamma} \right)_{M=0}$  is the incompressible effect of uniform dihedral obtained from Figure 4.3.1.3.

$K_{M_\Gamma}$  is the compressibility correction from Figure 4.3.1.4

The summary calculations of the wing contribution to  $C_{L_B}$  are shown in Table 4.3.1.1. For the ATLIT airplane

$$(C_{L_3})_w = -.000266 C_{L_w} - .00154 \quad (4.3.1.5)$$

#### 4.3.2 Effect of Fuselage on Wing Contribution to $C_{L_3}$

While the fuselage itself has negligible  $C_{L_3}$ , it does provide three wing-fuselage interference effects which are quite appreciable.

The first effect is caused by the vertical position of the wing. The flow about the fuselage in sideslip induces vertical flow components

which cause a change in the effective wing angle of attack. Without geometric dihedral, a high wing will produce a negative  $C_{l\beta}$ , a low wing will produce positive  $C_{l\beta}$ , and a mid wing will generally show no change. Equation 4.3.2.1 from Reference 3 is used to obtain this first interference effect.

$$(C_{l\beta})_{f(w)} \Gamma=0 = \frac{1.2\sqrt{A_w}}{57.3} \frac{z_w}{b_w} \frac{(h+w)}{b_w} \quad (4.3.2.1)$$

where

$z_w$  is the vertical distance from the axis of the equivalent circular fuselage to the quarter chord of the exposed wing panels, positive down, obtained from Figure 2.1.4.

$h$  is the height of the fuselage at the wing location, from Figure 4.3.2.1

$w$  is the width of the fuselage at the wing location, from Figure 4.3.2.1

For wings with geometric dihedral, the position of the wing with respect to the fuselage changes along the span. This is the second fuselage interference effect and is found by Equation 4.3.2.2 from Reference 3.

$$(C_{l\beta})_{f(\Gamma)} = -0.0005 \sqrt{A_w} \left[ \frac{(d_f)_w}{b_w} \right]^2 \Gamma \text{ per deg} \quad (4.3.2.2)$$

where

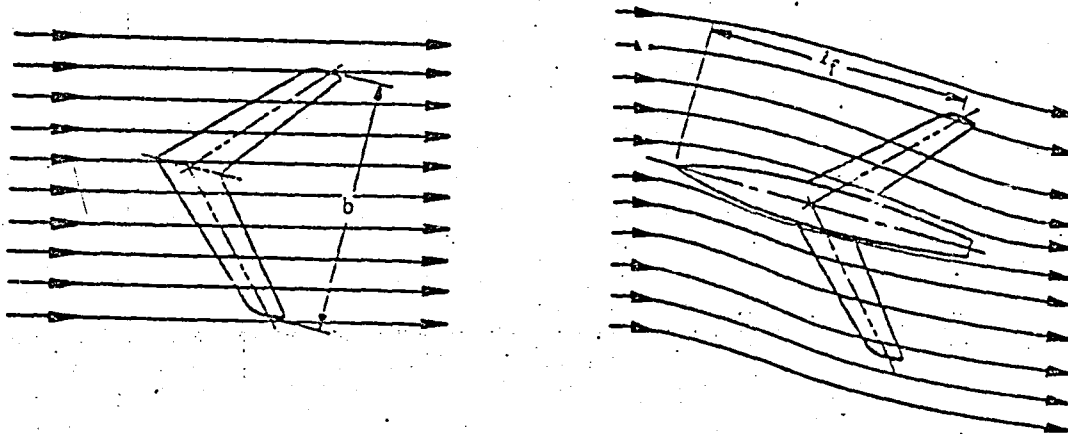
$(d_f)_w$  is the diameter of the equivalent circular fuselage at the wing, obtained from Figure 2.1.4.

$\Gamma$  is the geometric dihedral, in degrees, obtained from Figure 3.2.1.



The third fuselage interference effect is caused by the tendency of the fuselage to straighten the flow and therefore reduce the effective sideslip angle (see Sketch A). Reference 6 shows that this effect is a function of forebody length,  $l_f$ , wing span, and wing sweep. Reference 4 indicates that the parameter  $A_w/\cos\Lambda_{c/2}$  is also important. On the basis of Reference 6 this factor is considered zero for no sweep. For most general aviation airplanes this is probably a reasonable assumption until systematic studies are made on this effect.

Table 4.3.2.1 summarizes the fuselage interference contributions to  $C_{l_B}$  of the ATLIT airplane.



SKETCH A

#### 4.3.3 Vertical Tail Contribution to $C_{l_B}$

The vertical tail contributes to the airplane  $C_{l_3}$  by way of the rolling moment produced by its side force due to sideslip. Equation 4.3.3.1 from Reference 3 is used to find the vertical tail contribution.

$$(C_{l_B})_{v(wfh)} = -(C_{Y_B})_{v(wfh)} \frac{z_v \cos \alpha + l_v \sin \alpha}{b_w} \quad (4.3.3.1)$$

where

$(C_{Y_B})_{v(wfh)}$  is the vertical tail side force due to sideslip in the presence of the wing, fuselage, and horizontal tail, obtained from Table 4.1.4.1(c)

$z_v$  is the distance from the center of gravity to the vertical tail m.a.c. perpendicular to the X-body axis, from Figure 2.1.3.

$l_v$  is the distance from the center of gravity to the vertical tail m.a.c. parallel to the X-body axis, obtained from Figure 2.1.3.

Table 4.3.3.1 summarizes the vertical tail contribution of the ATLIT airplane.

#### 4.3.4 $C_{l_B}$ of the ATLIT Airplane

The total rolling moment due to sideslip of the complete airplane (propellers off) is

$$(C_{l_B})_{\text{prop off}} = (C_{l_B})_{w_{\Gamma=0}} + (C_{l_B})_{\Gamma} + (C_{l_B})_{f(w)_{\Gamma=0}} + (C_{l_B})_{f(\Gamma)} + (C_{l_B})_{v(wfh)} \quad (4.3.4.1)$$

Table 4.3.4.1 summarizes the contribution of each component and shows the  $C_{l_B}$  of the ATLIT airplane. In Figure 4.3.4.1 the predicted  $C_{l_B}$  is compared to the wind-tunnel data. Fairly good correlation is shown up to the non-linear lift range.

Table 4.3.1.1: Wing Contribution to  $C_{l_3}$   
(a) Zero Dihedral Component

Symbol	Description	Reference	Magnitude
$M$	Mach number	-	0.081
$C_{L_W}$	Wing lift coefficient	Figure 4.1.1.1	variable
$A_W$	Wing aspect ratio	Table 2.1.1	10.32
$\lambda_W$	Wing taper ratio	Table 2.1.1	0.5
$\left(\frac{C_{l_3}}{C_{L_W}}\right)_{M=0}$	Variation of $C_{l_3}$ with wing lift, per deg	Figure 4.3.1.1	-0.000226
$\bar{y}^*$	Spanwise position of the centroid of span loading as a fraction of semispan	Figure 4.3.1.2	0.42
Summary: $(C_{l_3})_{M=0} = -0.000226 C_{L_W}$ , per deg.			

(b) Dihedral Component

Symbol	Description	Reference	Magnitude
$\Gamma$	Wing dihedral angle, deg	Table 2.1.1	7
$\Lambda_c/2$	Wing half-chord sweep angle, deg	Table 2.1.1	0
$\left(\frac{C_{L_3}}{\Gamma}\right)_{M=0}$	Low speed effect of dihedral	Figure 4.3.1.3	-0.00022
$K_{M_\Gamma}$	Compressibility correction factor	Figure 4.3.1.4	1.0
Summary: $(C_{L_3})_\Gamma = -0.00154$ , per deg.			

Table 4.3.2.1: Fuselage Effect on Wing  $C_{L_3}$

Symbol	Description	Reference	Magnitude
$A_w$	Wing aspect ratio	Table 2.1.1	10.32
$b_w$	Wing span, m (ft)	Table 2.1.1	12.19 (40.0)
$z_w$	Vertical distance of wing root chord below centerline of the equivalent circular fuselage, m (ft)	Figure 2.1.4	0.194 (0.638)
$(d_f)_w = h = w$	Diameter of equivalent circular fuselage at wing, m (ft)	Figure 2.1.4	1.22 (4.0)
$\Gamma$	Wing dihedral angle, deg	Table 2.1.1	7
Summary: $(C_{L_3})_{f(w), \Gamma=0} + (C_{L_3})_{f(\Gamma)} = 0.000219 - 0.000112 = 0.000107$ , per deg			

Table 4.3.3.1: Vertical Tail Contribution to  $C_{l_3}$

Symbol	Description	Reference	Magnitude
$(C_{Y_3})_{v(wfn)}$	Vertical tail contribution to $C_{Y_3}$	Table 4.1.3.1	-0.0056
$z_v$	Distance perpendicular to X-body axis from c.g. to quarter-chord of mean aerodynamic chord of vertical tail, m (ft)	Figure 2.1.5	4.50 (14.75)
$l_v$	Distance along X-body axis from c.g. to quarter chord of mean aerodynamic chord of vertical tail, m (ft)	Figure 2.1.5	1.22 (4.0)
$b_v$	Wing span, m (ft)	Table 2.1.1	12.19 (40.0)
Summary: $(C_{l_3})_{v(wfn)} = 0.002065 \sin \alpha - 0.00056 \cos \alpha$ , per deg			

Table 4.3.4.1:  $C_{L_3}$  of the ATLIT Airplane

$\alpha$ , deg	$C_{L_w}$ Fig. 4.1.1.1	$(C_{L_3})_{w_r=0}$ Table 4.3.1.1 (a)	$(C_{L_3})_r$ Table 4.3.1.1 (b)	$(C_{L_3})_{f(w)}$ Table 4.3.2.1	$(C_{L_3})_{v(wfn)}$ Table 4.3.3.1	$(C_{L_3})_{prop\ off}$ Eq. 4.1.4.1
-4	-0.0533	0.000012	-0.00154	0.000107	-0.000703	-0.002124
-2	0.1243	-0.000028	-0.00154	0.000107	-0.000632	-0.002093
0	0.3019	-0.000068	-0.00154	0.000107	-0.000560	-0.002061
2	0.4795	-0.000108	-0.00154	0.000107	-0.000488	-0.002029
4	0.6571	-0.000149	-0.00154	0.000107	-0.000415	-0.001997
15.9	1.494	-0.000338	-0.00154	0.000107	0.000027	-0.001744

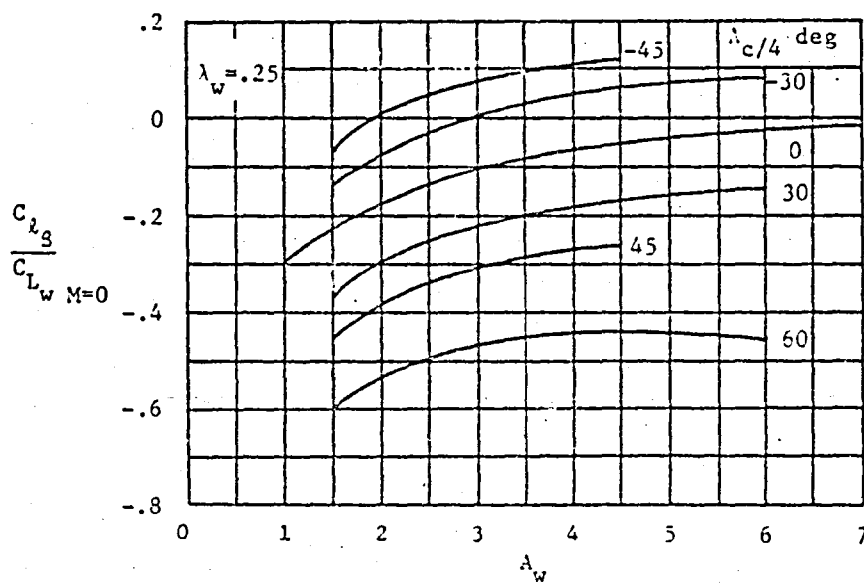
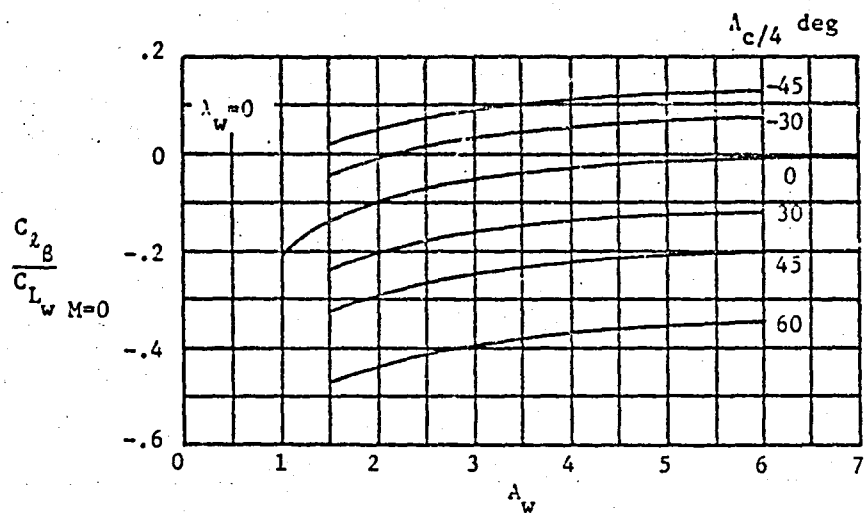


Figure 4.3.1.1: Incompressible variation of  $C_{l_g}$  with wing lift (Reference 3)



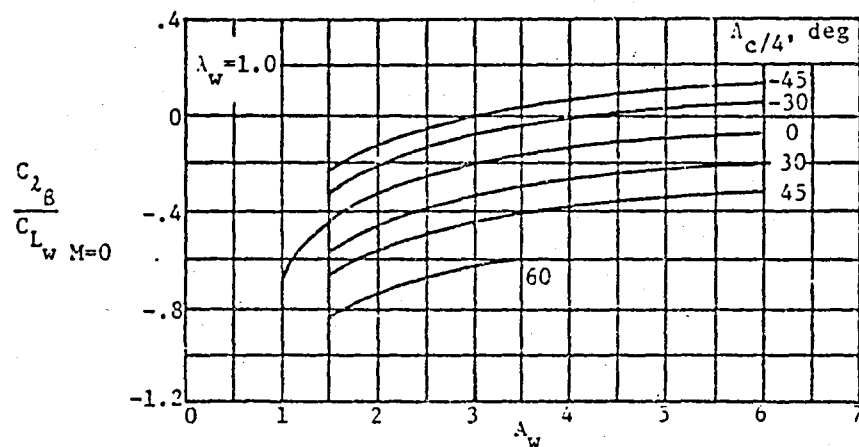
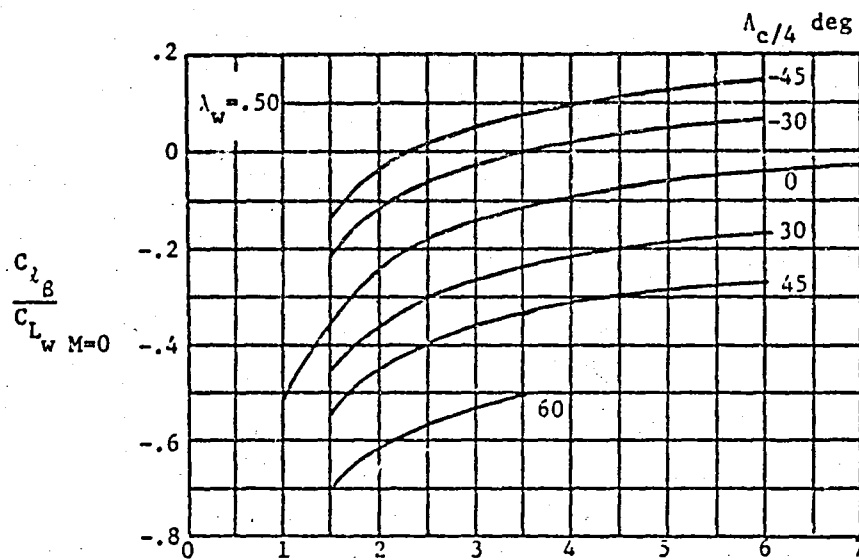


Figure 4.3.1.1: (continued)

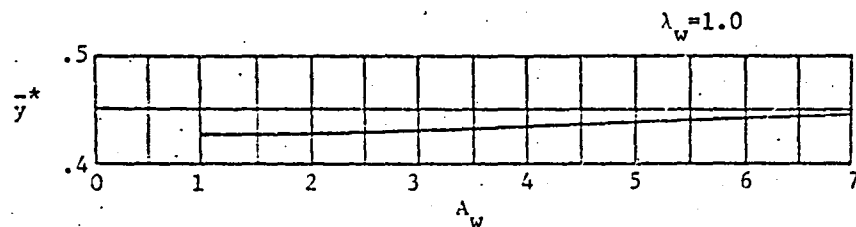
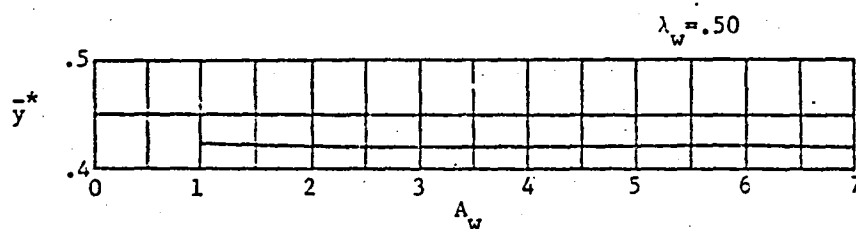
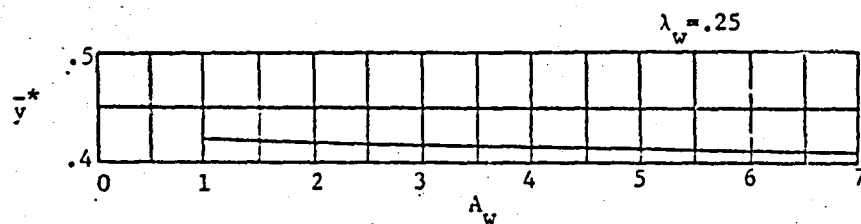
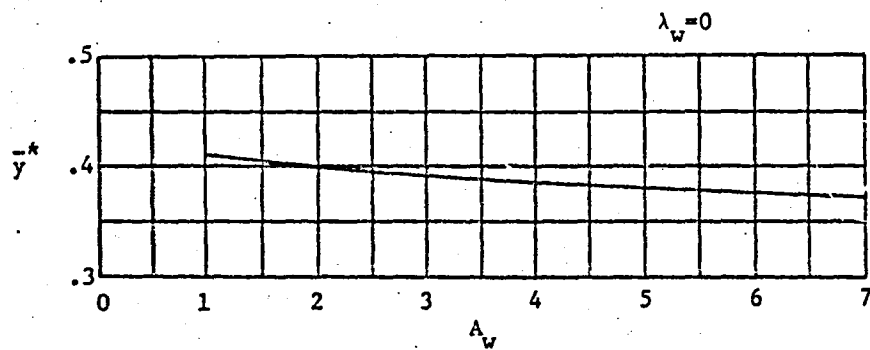


Figure 4.3.1.2: Spanwise location of centroid of angle of attack of loading for unswept wings (Reference 3)

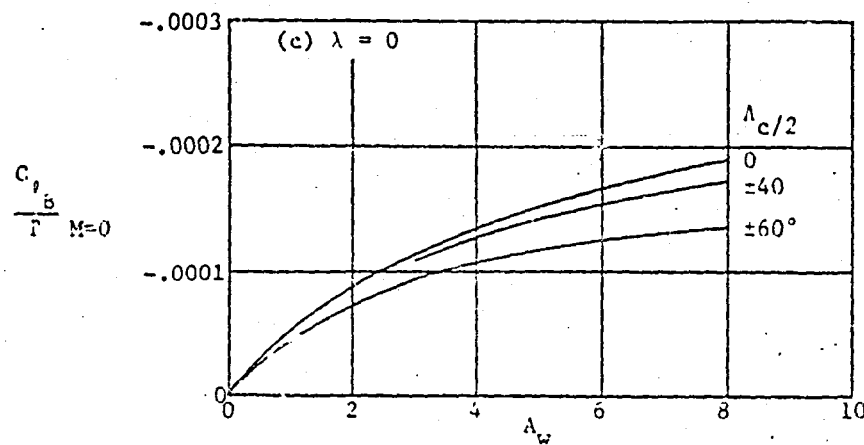
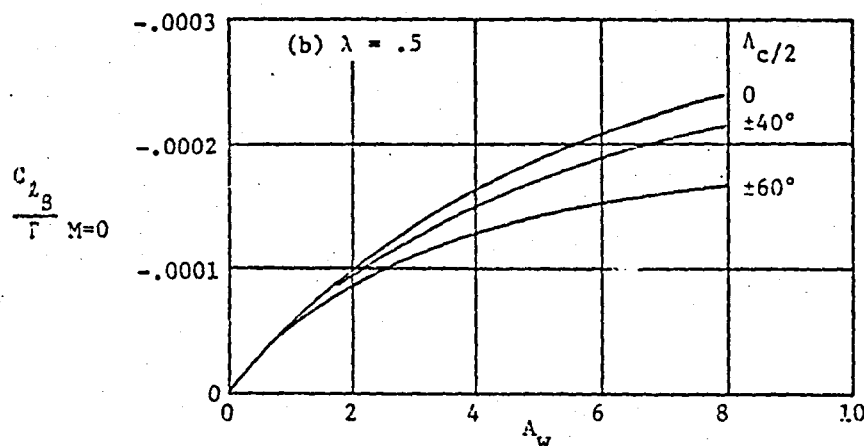
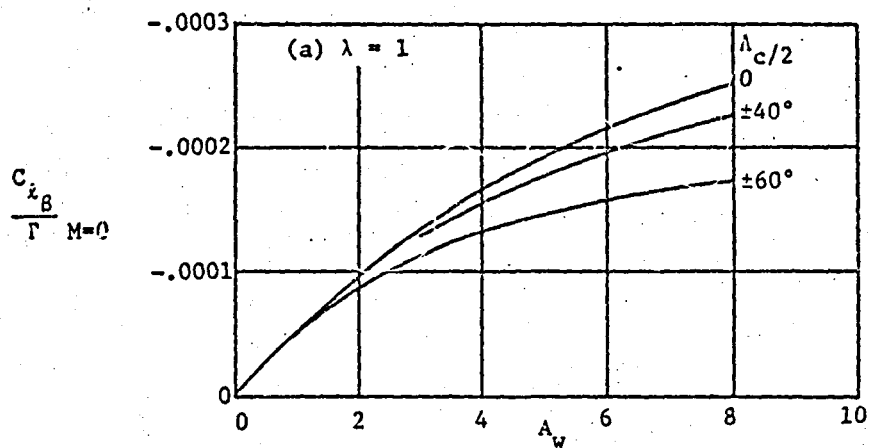


Figure 4.3.1.3: Effect of Uniform Geometric Dihedral on Wing  $C_{iB}$  (Reference 3)

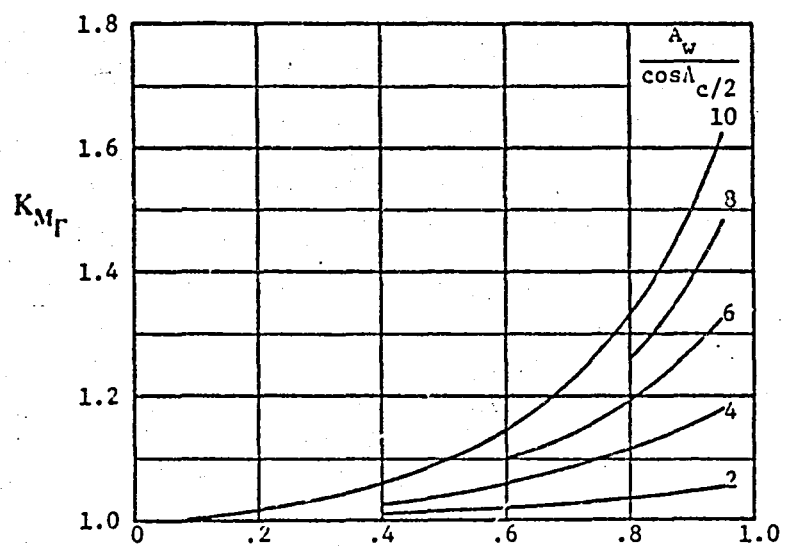


Figure 4.3.1.4: Compressibility correction for dihedral contribution to  $C_{L\beta}$   
(Reference 3)

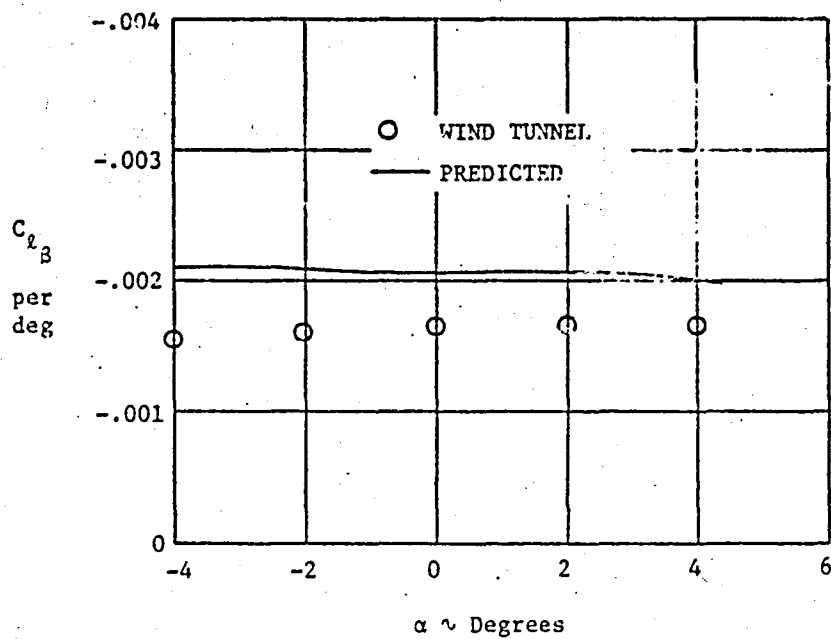


Figure 4.3.4.1: Comparison of predicted  $C_{L_B}$  with full scale wind tunnel data (propellers removed,  $N_{Re} = 2.3$  million)

#### 4.4 Spoiler Derivatives, $C_{l_{\delta s}}$ , $C_{n_{\delta s}}$

In order to free the entire trailing edge of the wing for high-lift devices, spoilers were used to provide roll control. The spoilers used are vented, gapped, upper-surface roll-control spoilers. For a discussion of the design decisions made in developing the spoiler system, see Reference 5. Figure 4.4.1 shows the spoiler installation.

##### 4.4.1 Rolling Moment Due to Spoilers, $C_{l_{\delta s}}$

The method used in obtaining the spoiler derivative  $C_{l_{\delta s}}$  is taken from Reference 4 and is based on a simplified lifting surface theory. This method is valid only for attached flow conditions. Reference 5 outlines some of the characteristics of spoiler in the stall region.

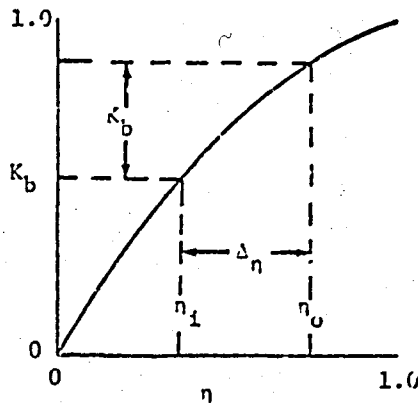
The rolling moment produced by deflecting one spoiler is given in Equation 4.4.1.1.

$$C_l = \frac{C'_{l_{\delta s}}}{2} \Delta\alpha'_s \quad (4.4.1.1)$$

where

$\Delta\alpha'_s$  is the spoiler lift effectiveness in terms of the change in zero-lift angle of attack, obtained from Figure 4.4.1.1

$C'_{l_{\delta s}}$  is the rolling moment effectiveness of two full chord controls deflected antisymmetrically, obtained from Figure 4.4.1.2 based on the effective inboard and outboard ends of the control. For partial span controls, this parameter is obtained as shown in Sketch A.



Sketch A

The effective inboard and outboard spoiler locations are given in Equation 4.4.1.2 from Reference 4.

$$\eta_{i\text{eff}} = \eta_i + \Delta \eta_i \quad (4.4.1.2)$$

$$\eta_{o\text{eff}} = \eta_o + \Delta \eta_o$$

where

$\eta_i, \eta_o$  are the actual inboard and outboard ends of the spoiler given by  $\eta = \frac{y}{b/2}$

$\Delta \eta_i, \Delta \eta_o$  are the effective increments of the spoiler locations due to the spanwise flow of the spoiler wake, obtained from Equation 4.4.1.3.

$$\left. \begin{aligned} \Delta \eta_i &= \frac{4(1 - \frac{x'_s}{c})}{A(1 + \lambda)} [1 - (1 - \lambda)\eta_i] \frac{\cos \lambda_{TE} \sin \theta}{\cos(\lambda_{TE} + \theta)} \\ \Delta \eta_o &= \frac{4(1 - \frac{x'_s}{c})}{A(1 + \lambda)} [1 - (1 - \lambda)\eta_o] \frac{\cos \lambda_{TF} \sin \theta}{\cos(\lambda_{TF} + \theta)} \end{aligned} \right\} \quad (4.4.1.3)$$

where

$\theta$  is determined from Figure 4.4.1.3 as a function of spoiler sweepback given in Equation 4.4.1.4.

$$\tan \Lambda_s = \tan \Lambda_{c/4} - \frac{4[0.75 - (1 - \frac{x'_s}{c})]}{A} (\frac{1-\lambda}{1+\lambda}) \quad (4.4.1.4)$$

Using the above procedure, the spoiler rolling moment can be determined for various spoiler deflections and  $C_{l_{\delta_s}}$  then found.

For the ATLIT spoiler deflections of 10, 20, 40, and 60 degrees were used. Table 4.4.1.1 summarizes the calculations for the ATLIT airplane. The predicted values are presented in Figure 4.4.1.4 and compared with wind tunnel data.



#### 4.4.2 Yawing Moment Due to Spoilers, $C_{n\delta_s}$

The yawing moment due to spoilers comes from two effects. The first is due to a decrease in induced drag on one wing; the second is an increase in profile drag on the wing with the deflected spoiler. The method used, taken from Reference 4, includes both effects.

For unswept wing at low angle of attack, the yawing moment is found directly from Figure 4.4.2.1. As in the case of the rolling moment,  $C_n$  should be computed for several spoiler deflection angles and  $C_{n\delta_s}$  then found as a function of  $\delta_s$ .

Table 4.4.2.1 summarizes the ATLIT  $C_{n\delta_s}$  calculations. Figure 4.4.2.2 shows a comparison with wind-tunnel results.

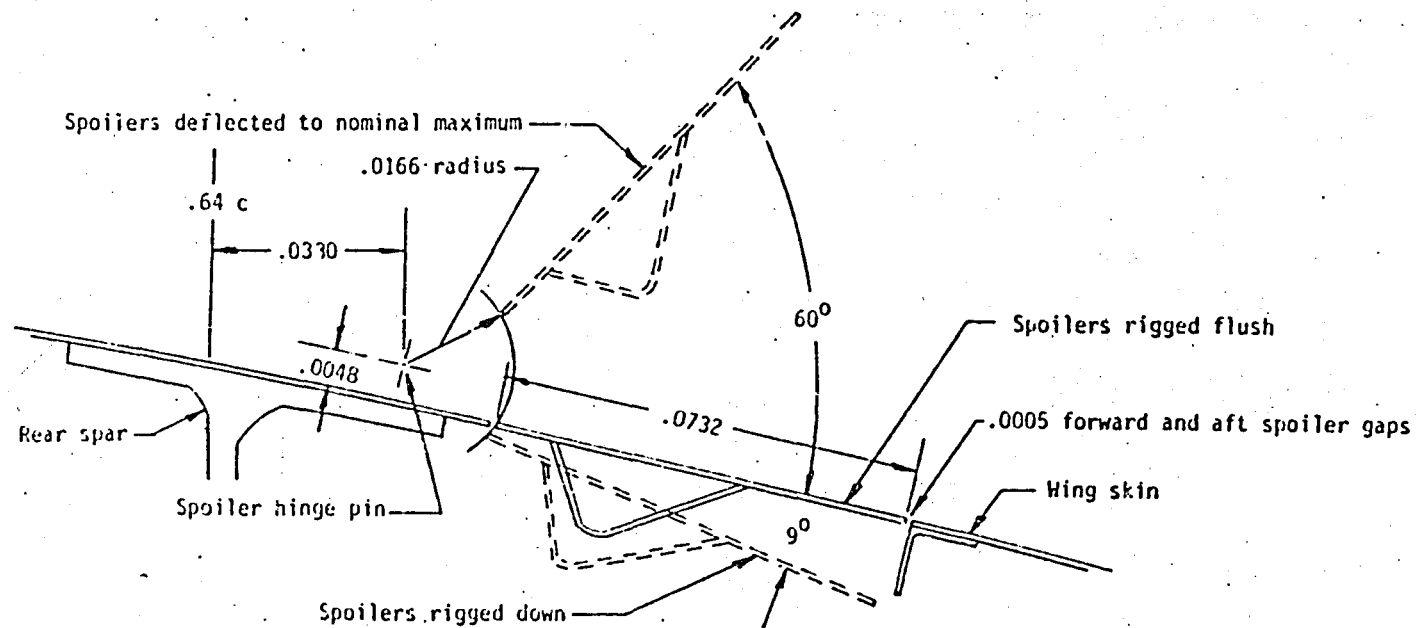


Figure 4.4.1: Spoiler Installation (Reference 7)

Table 4.4.1.1: Spoiler Effectiveness of the ATLIT Airplane

Symbol	Description	Reference	Magnitude
$\Delta\alpha'_s$	Change in zero-lift angle of attack due to spoiler deflection, radians	Figure 4.4.1.1	variable
$C_{l_s}$	Rolling moment effectiveness of two full chord spoilers, per rad	Figure 4.4.1.2	0.7507
$\eta_i, \eta_o$	Inboard and outboard nondimensional spoiler locations, $\eta = 2y/b$	Figure 2.1	0.421, 0.95
$\eta_{i,eff}, \eta_{o,eff}$	Effective spoiler locations	Equation 4.4.1.2	0.410, 0.943
$\theta$	Spanwise deflection of spoiler wake, deg	Figure 4.4.1.3	-7
$\Lambda_s$	Spoiler sweep angle, deg	Equation 4.4.1.3	-1.5
$\delta_s$	Spoiler deflection angle, deg		variable
$\frac{h_s}{c}$	Distance from airfoil mean line to end of spoiler	Figure 4.4.1	variable
$\frac{x_s}{c}$	Distance from airfoil leading edge to end of spoiler	Figure 4.4.1	variable

$\delta_s$ , deg	$\frac{x_s}{c}$	$\frac{h_s}{c}$	$\Delta\alpha'_s$	$C_{l_s}$
10	0.771	0.054	0.056	0.021
20	0.768	0.066	0.087	0.033
40	0.755	0.090	0.108	0.041
60	0.736	0.110	0.128	0.048

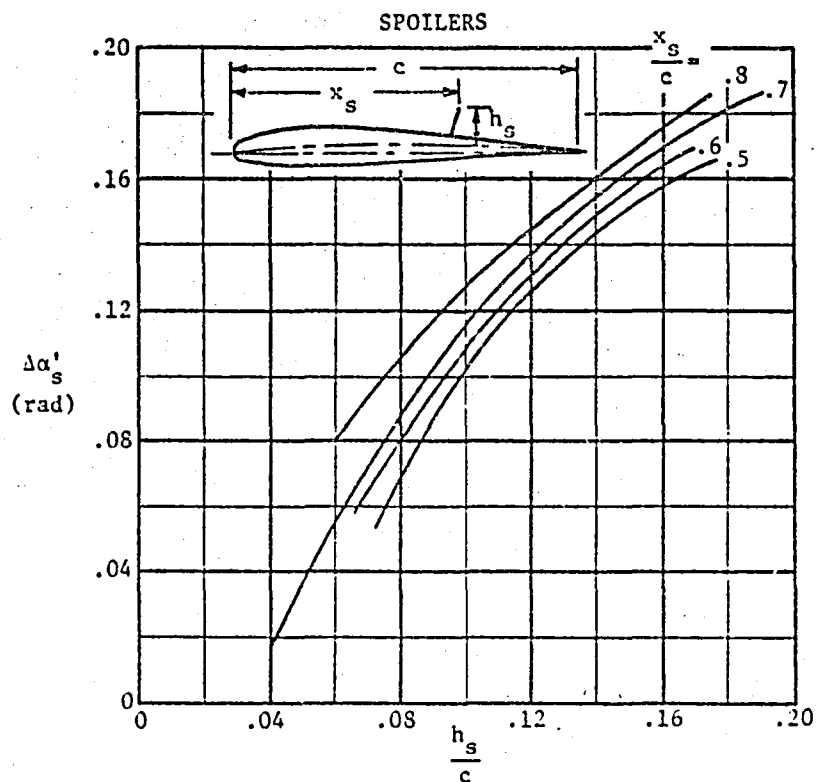


Figure 4.4.1.1: Spoiler lift effectiveness  
(Reference 4)

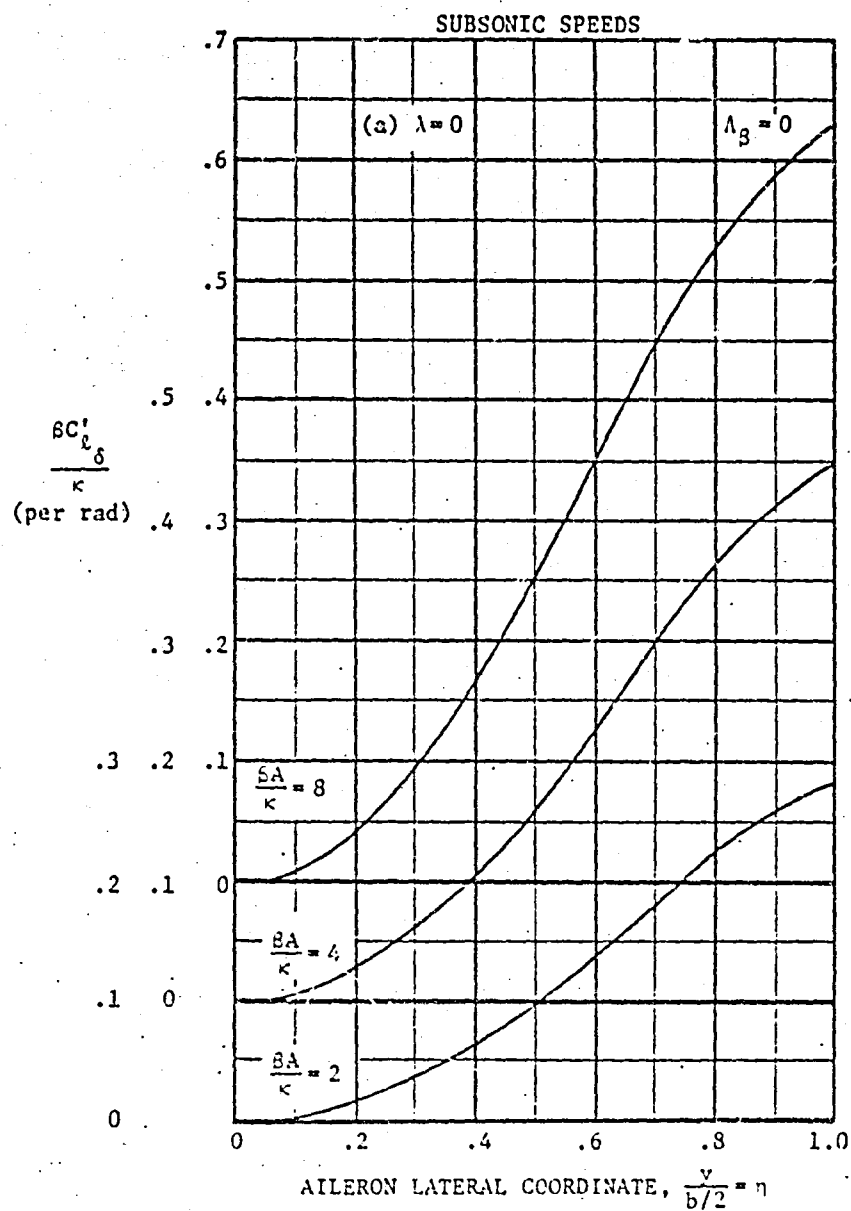


Figure 4.4.1.2(a): Rolling moment effectiveness  
(Reference 4)

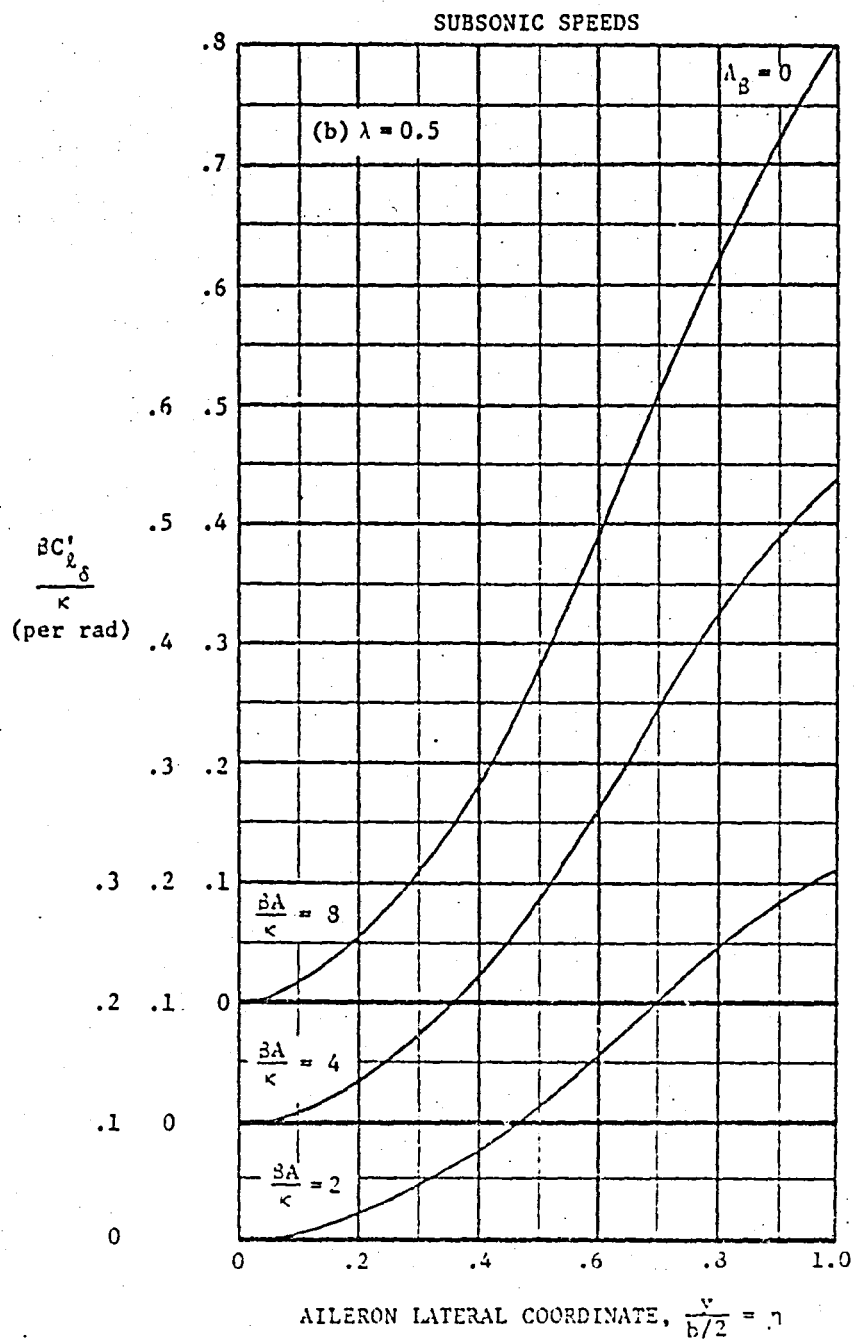


Figure 4.4.1.2(b): Rolling moment effectiveness  
(Reference 4)

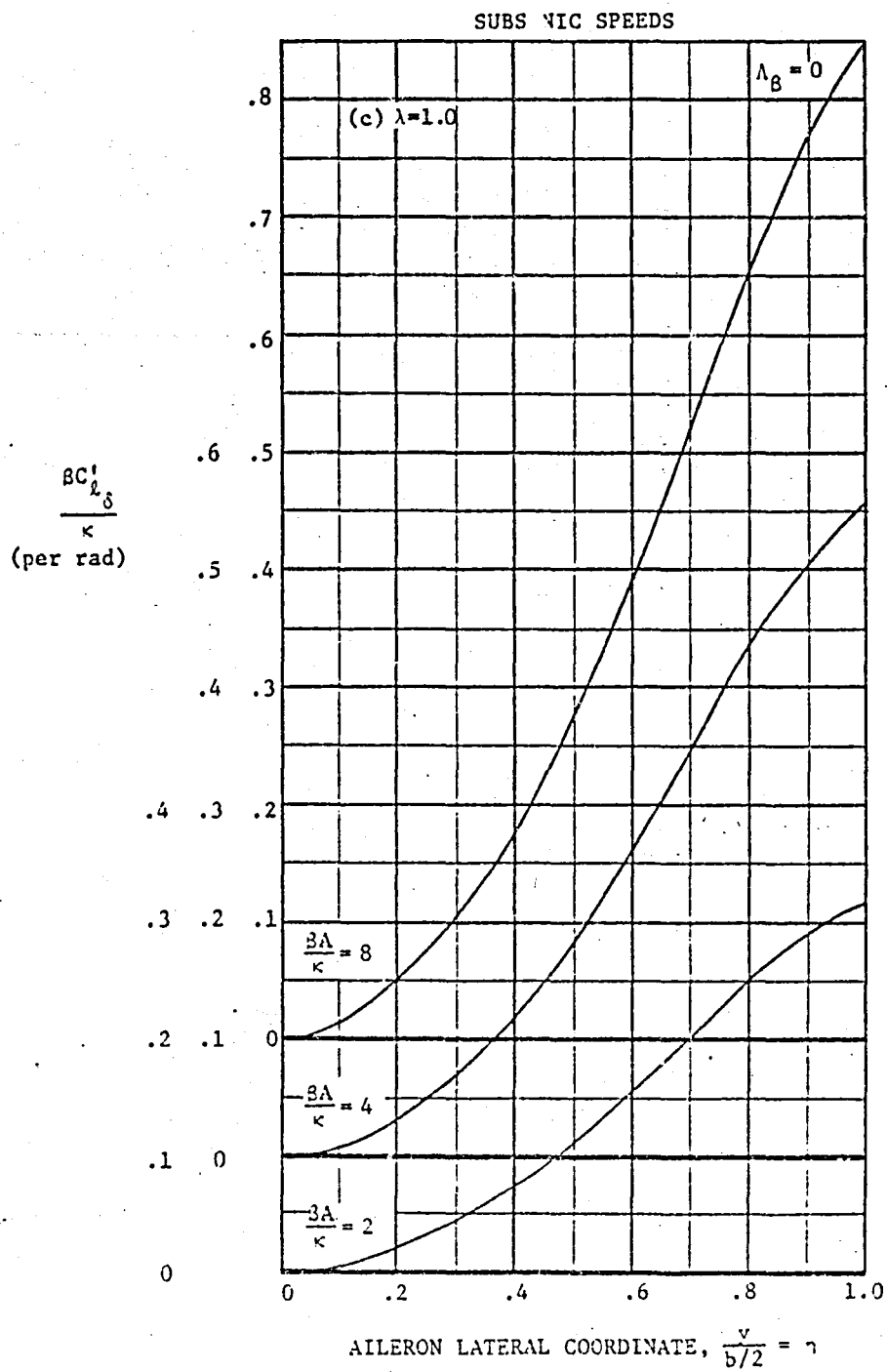


Figure 4.4.1.2(c): Rolling moment effectiveness-  
(Reference 4)

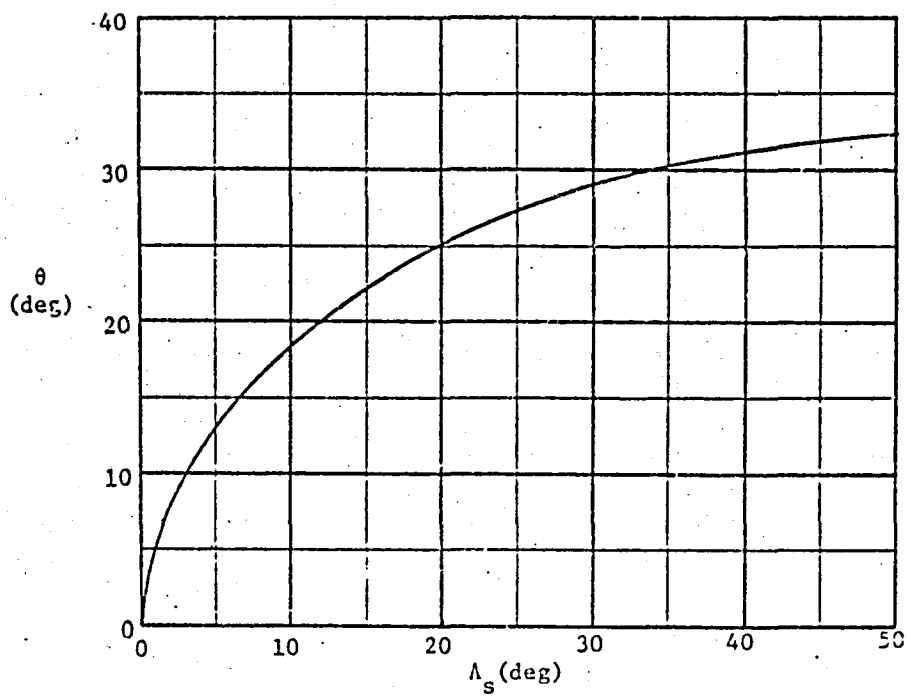


Figure 4.4.1.3: Variation of  $\theta$  with spoiler sweep (Reference 4)



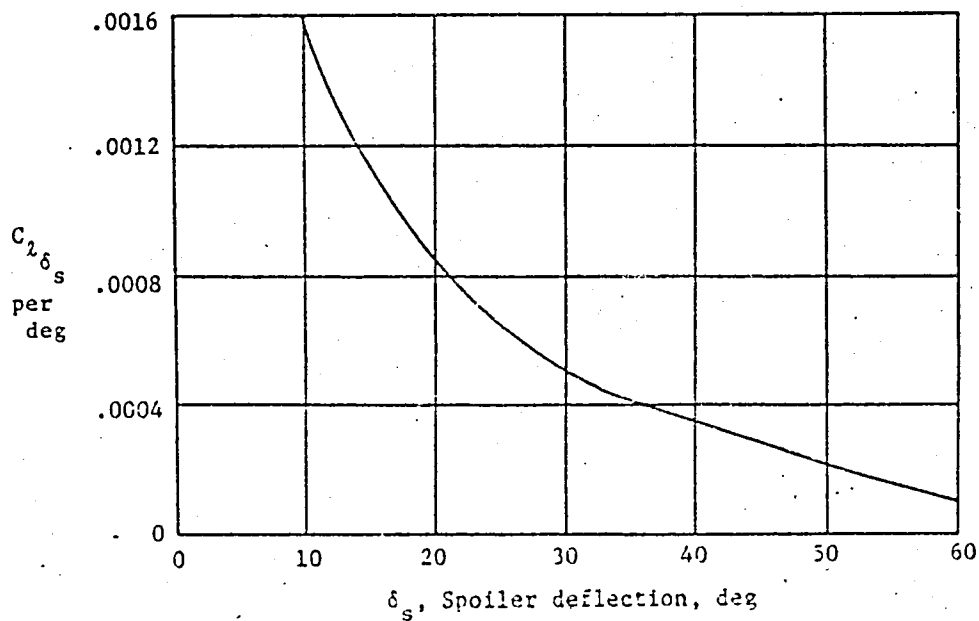
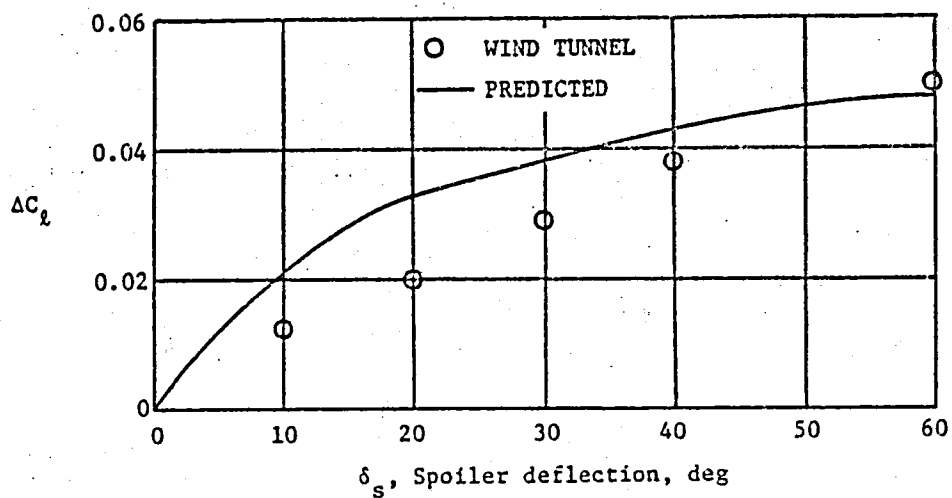


Figure 4.4.1.4 Comparison of predicted spoiler effectiveness with full-scale wind tunnel data (Power off;  $\alpha = 0^\circ$ )

Table 4.4.2.1: Yawing Moment Due to Spoilers on the ATLIT Airplane

Symbol	Description	Reference	Magnitude
$b_s$	Spoiler span, m (ft)	Table 2.1	3.22 (10.58)
$\lambda_w$	Wing taper ratio	Table 2.1	0.5
$\frac{x_s}{c}$	Distance from airfoil leading edge to end of spoiler	Figure 4.4.1	variable
$\frac{h_s}{c}$	Distance from airfoil mean line to end of spoiler	Figure 4.4.1	variable

$\delta_s$ , deg	$\frac{x_s}{c}$	$\frac{h_s}{c}$	$C_n$
10	0.771	0.054	0.0013
20	0.768	0.066	0.0027
40	0.755	0.090	0.0061
60	0.736	0.110	0.0127

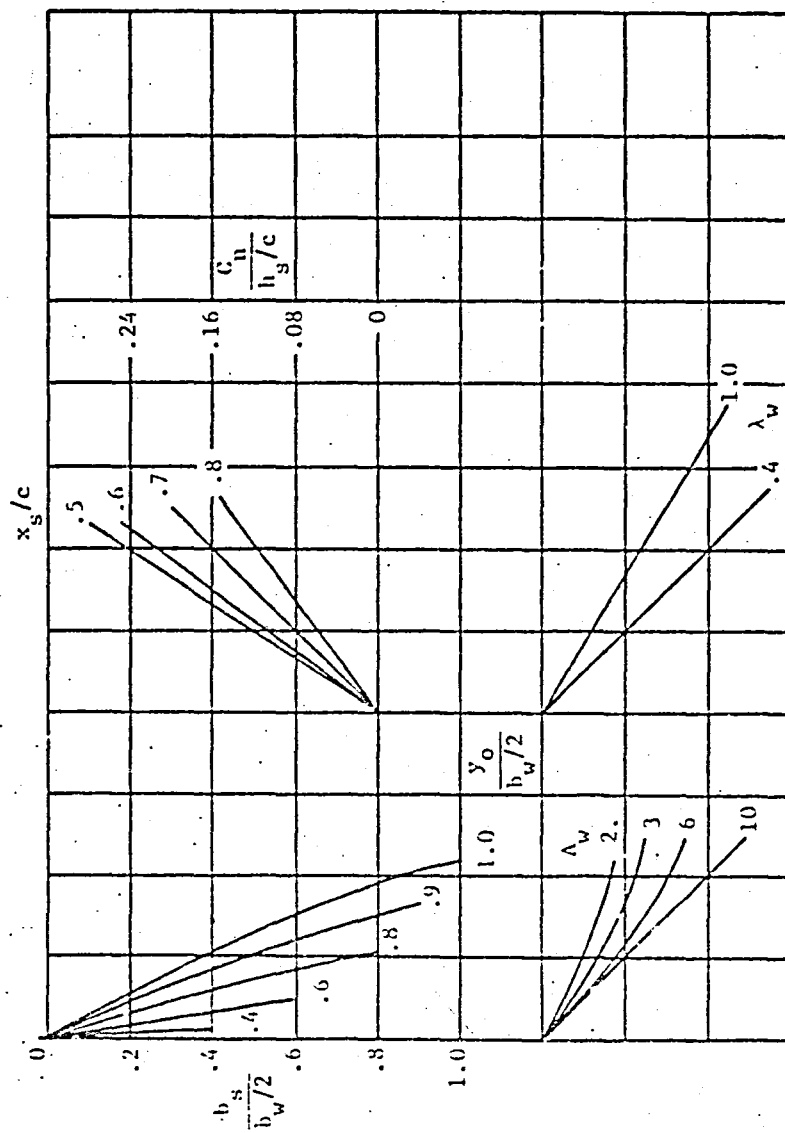


Figure 4.4.2.1: Yawing moment due to spoiler deflection (Reference 4)

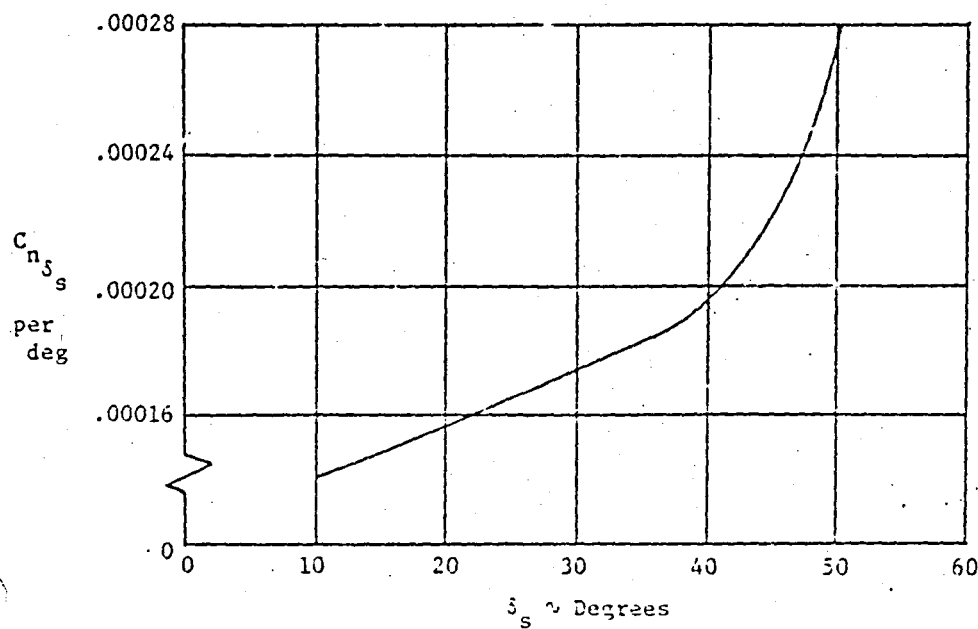
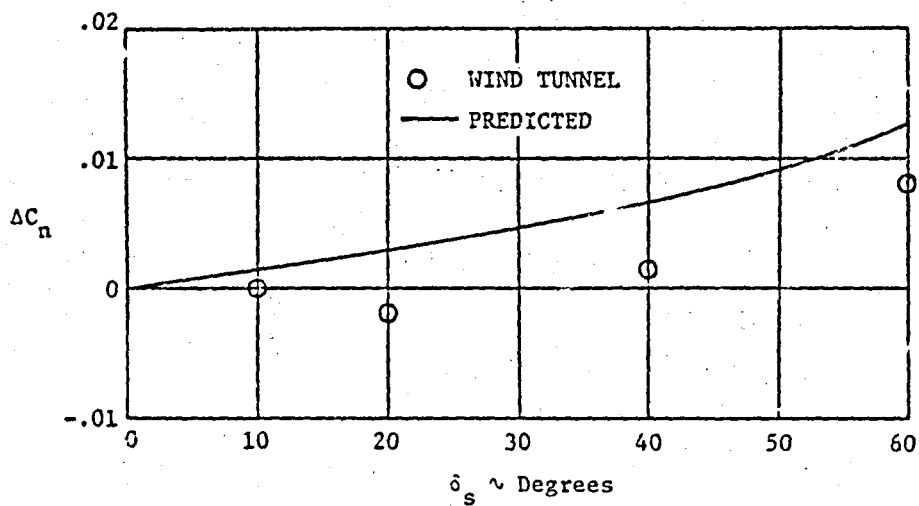


Figure 4.4.2.2 Comparison of predicted yaw due to spoilers with wind tunnel data (propellers removed,  $\alpha=0^\circ$ ).

#### 4.5 Rudder Control Derivatives, $C_{Y_{\delta_R}}, C_{n_{\delta_R}}, C_{l_{\delta_R}}$

The rudder derivatives are found by assuming that the rudder is a simple flap. The side force is determined directly and the yawing and rolling moments are found by considering the side force acting through the appropriate moment arm.

##### 4.5.1 Side Force Due to Rudder

The side force due to rudder deflection is found by Equation 4.5.1.1 from Reference 3.

$$C_{Y_{\delta_R}} = C_{l_{\delta_R}} \frac{(C'_{L_{\alpha}})_v}{(C_{l_{\alpha}})_v} \left[ \frac{(\alpha_{\delta_r}) C_L}{(\alpha_{\delta_r}) C_l} \right]_v K_b \quad (4.5.1.1)$$

where

$(C'_{L_{\alpha}})_v$  is the effective lift curve slope of the vertical tail, obtained from Equation 4.5.1.2. This is the same as  $-(C_{Y_3})_v(wfh)$  from Equation 4.1.4.5 with  $\partial c / \partial \beta = 0$ .

$$(C'_{L_{\alpha}})_v = K'_1 (C_{L_{\alpha}})_v(fh) \frac{\bar{q}_v S_v}{\bar{q}_w S_w} \quad (4.5.1.2)$$

$(C_{l_{\alpha}})_v$  is the section lift curve slope of the vertical tail from Reference 1, Section 4.1

$K_b$  is the rudder span factor obtained from Figure 4.5.1.1

$\left[ \frac{(\alpha_{\delta_r}) C_L}{(\alpha_{\delta_r}) C_l} \right]_v$  is the rudder chord factor obtained from Figure 4.5.1.2

as a function of effective vertical tail aspect ratio and the ratio of rudder chord to vertical tail chord  $\frac{c_r}{c_v}$ . If  $\frac{c_r}{c_v}$  is not constant, an average value should be used.

$C_{l_{\delta}}$  is the section lift effectiveness of the rudder obtained from Equation 4.5.1.3

$$C_{l_{\delta_R}} = \frac{1}{B_1} \frac{C_{l_{\delta}}}{(C_{l_{\delta}})_{\text{theory}}} (C_{l_{\delta}})_{\text{theory}} K' \quad (4.5.1.3)$$

where

$(C_{l_{\delta}})_{\text{theory}}$  is the theoretical value of section lift effectiveness obtained from Figure 4.5.1.3

$\frac{C_{l_{\delta}}}{(C_{l_{\delta}})_{\text{theory}}}$  is a correction factor obtained from Figure 4.5.1.4 as a function of  $c_r/c_v$  and

$\frac{C_{l_{\alpha}}}{(C_{l_{\alpha}})_{\text{theory}}}$ , where  $(C_{l_{\alpha}})_{\text{theory}}$  is obtained from Equation 4.5.1.3 as a function of thickness ratio and trailing edge angle  $\phi_{TE}$

$$(C_{l_{\alpha}})_{\text{theory}} = 6.20 + 4.7(t/c)(1 + 0.00375 \phi_{TE}) \text{ per radian} \quad (4.5.1.3)$$

$K'$  is a correction factor for reduced effectiveness at high deflection angles, obtained from Figure 4.5.1.5.

$$B_1 = \sqrt{1 - M^2}$$

The summary calculations for  $C_{Y_{\delta_R}}$  of the ATLIT airplane are shown in Table 4.5.1.1. Figure 4.5.1.6 compares the calculations with full-scale wind-tunnel results for power off.

#### 4.5.2 Rolling Moment Due to Rudder, $C_{l_{\delta_R}}$

The rolling moment due to rudder deflection is given by Equation 4.5.2.1 from Reference 3.

$$C_{l\delta_R} = -C_{Y\delta_R} \frac{l'_V \cos \alpha + z'_V \sin \alpha}{b_w} \quad (4.5.2.1)$$

where

$l'_V$  is the distance parallel to the X-body axis from the center of gravity to the quarter chord of the mean aerodynamic chord of the portion of the vertical tail spanned by the rudder.

$z'_V$  is the distance parallel to the Z-body axis from the center of gravity to the quarter chord of the mean aerodynamic chord of the portion of the vertical tail spanned by the rudder.

Note that for full span rudders,  $l'_V$  and  $z'_V$  are the same as  $l_V$  and  $z_V$  defined in Section 4.2.4.

Table 4.5.2.1 summarizes the ATLIT airplane's  $C_{l\delta_R}$ . Figure 4.5.2.1 shows the comparison with full-scale wind-tunnel results.

#### 4.5.3 Yawing Moment Due to Rudder, $C_{n\delta_R}$

The yawing moment due to rudder deflection is given by Equation 4.5.3.1 from Reference 3.

$$C_{n\delta_R} = -C_{Y\delta_R} \frac{l'_V \cos \alpha - z'_V \sin \alpha}{b_w} \quad (4.5.3.1)$$

where  $l'_V$  and  $z'_V$  are defined in Section 4.5.2.

The summary calculations for  $C_{n\delta_R}$  of the ATLIT airplane are shown in Table 4.5.3.1. In Figure 4.5.3.1 the predicted values are compared with wind-tunnel results.

Table 4.5.1.1:  $C_{Y_{\delta_r}}$  for the ATLIT Airplane

Symbol	Description	Reference	Magnitude
$S_v$	Vertical tail area	Table 2.1.1	1.85 (19.9)
$S_w$	Wing area	Table 2.1.1	14.40 (155.9)
$C_{L_{\alpha_v}}$	Effective lift curve slope of the vertical tail, per rad	Equation 4.5.1.2	0.2905
$(C_{l_{\alpha}})_v$	Section lift curve slope of the vertical tail, per rad	Table 4.2.4.1 Reference 1	6.245
$K_b$	Rudder span factor	Figure 4.5.1.1	1
$\left[ \frac{(a_{\delta_r}) C_{L_r}}{(a_{\delta_r}) C_{L_v}} \right]_v$	Rudder chord factor	Figure 4.5.1.2	1.13
$\lambda_v$	Vertical tail taper ratio	Table 2.1.1	0.459
$\frac{c_r}{c_v}$	Ratio of rudder chord to vertical tail chord	Table 2.1	= 0.20
$B_1$	$\sqrt{1-M^2}$	-	0.997
$\frac{C_{l_{\delta}}}{(C_{l_{\delta}})_{\text{theory}}}$	Correction factor for lift effectiveness	Figure 4.5.1.4	0.88
$(C_{l_{\delta}})_{\text{theory}}$	Theoretical section-lift effectiveness, per rad	Equation 4.5.1.3	3.60
$K'$	Deflection angle correction factor	Figure 4.5.1.5	1.0
$C_{l_{\delta_r}}$	Section lift effectiveness of rudder, per rad	Equation 4.5.1.3	3.18
Summary: $C_{Y_{\delta_r}} = 0.00239 \text{ per deg}$			



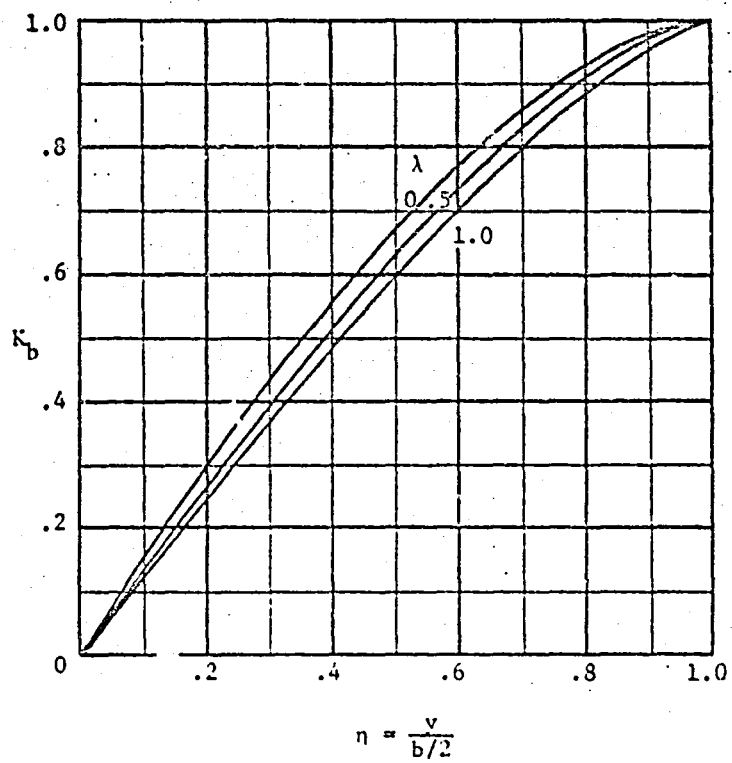


Figure 4.5.1.1: Rudder span factor (Reference 3)

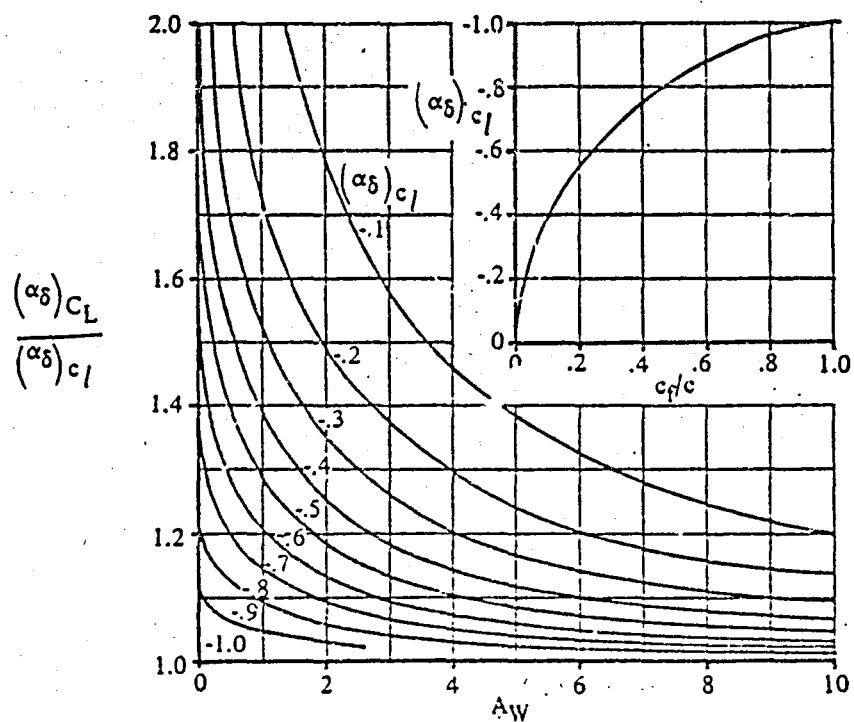


Figure 4.5.1.2: Rudder chord factor (Reference 3)

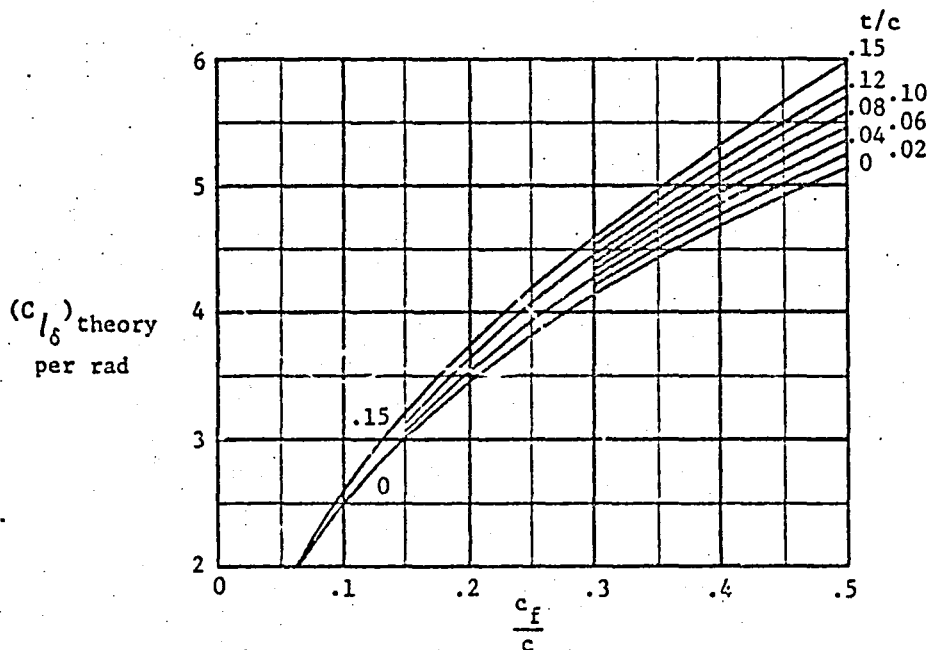


Figure 4.5.1.3: Theoretical section lift effectiveness (Reference 3)

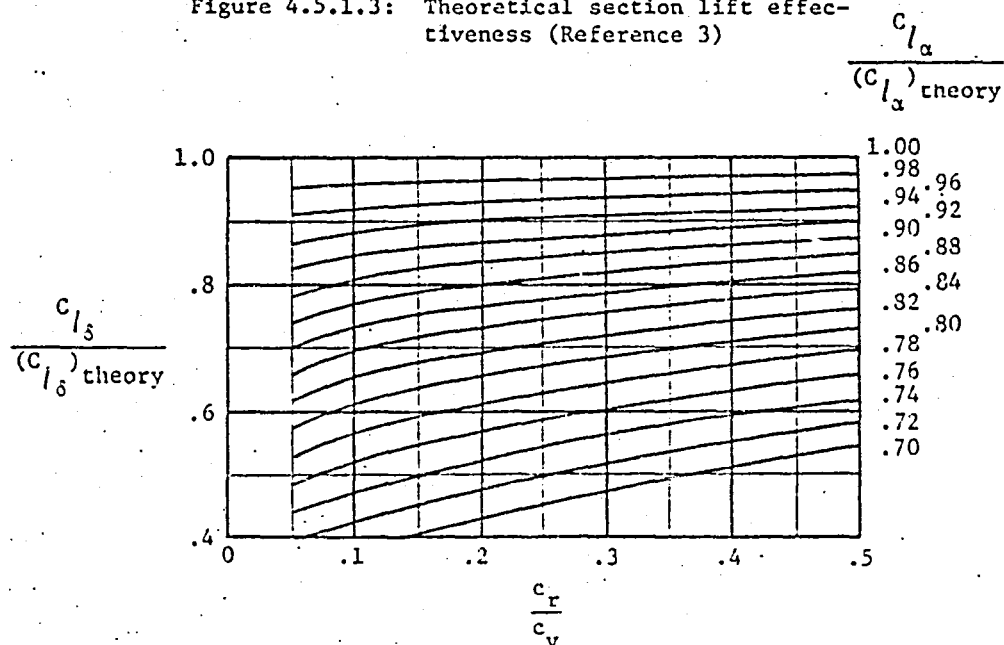


Figure 4.5.1.4: Empirical lift effectiveness correction (Reference 3)

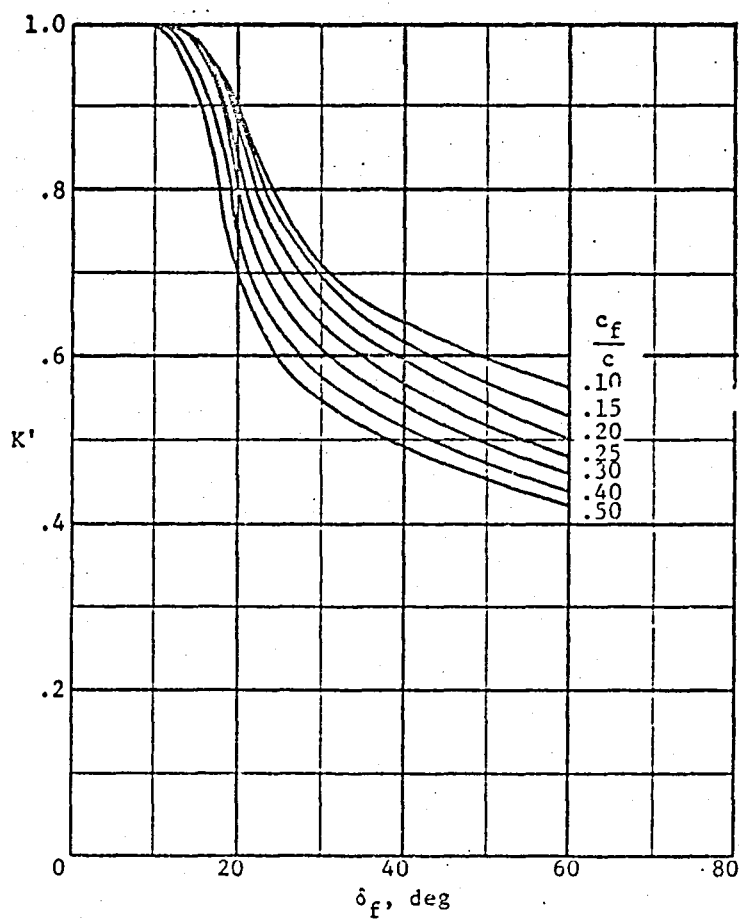


Figure 4.5.1.5: Empirical correction for high deflection angles (Reference 3)

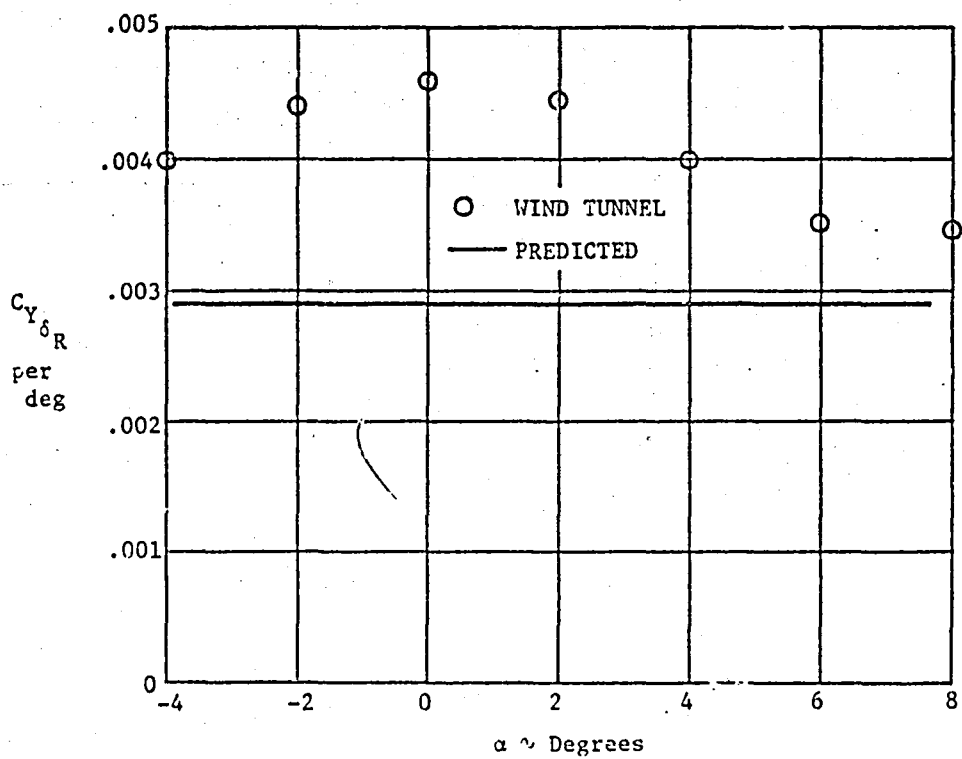


Figure 4.5.1.6: Comparison of predicted  $C_{Y_{\delta_R}}$  with wind tunnel data

( $\delta_R = \pm 10^\circ$ ,  $\beta = 0^\circ$ )

Table 4.5.2.1:  $C_{l_{\delta r}}$  of the ATLIT Airplane

Symbol	Description	Reference	Magnitude
$C_{Y_{\delta r}}$	Change in side force with rudder deflection, per deg	Table 4.5.1.1	0.00289
$l'_v$	Distance parallel to the X-body axis from the c.g. to the quarter-chord of the mean aerodynamic chord of the portion of the vertical tail spanned by the rudder, m (ft)	Figure 2.1.5	4.50 (14.75)
$z'_v$	Distance parallel to the Z-body axis from the c.g. to the quarter chord of the mean aerodynamic chord of the portion of vertical tail spanned by the rudder, m (ft)	Figure 2.1.5	-1.22 (-4.0)
$b_w$	Wing span, m (ft)	Table 2.1.1	12.19 (40.0)

$\alpha$ , deg	$C_{l_{\delta r}}$ per deg
-4	0.00036
-2	0.00033
0	0.00029
2	0.00025
4	0.00021
8	0.00014
15.9	-0.00001

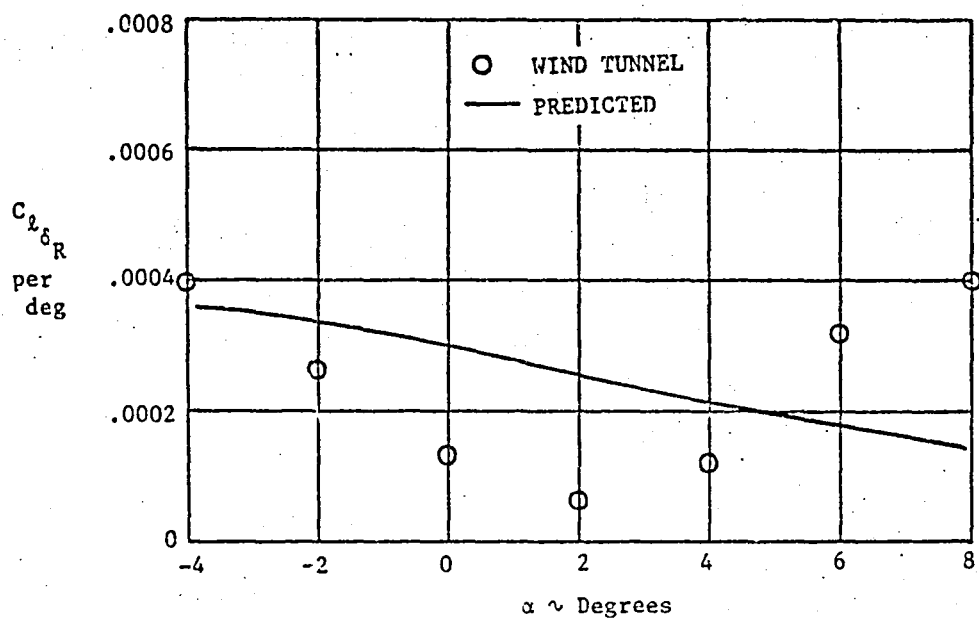


Figure 4.5.2.1: Comparison of predicted  $C_{l\delta_R}$  with wind tunnel data  
 $(\beta = 0^\circ, \delta_R = \pm 10^\circ)$

Table 4.5.3.1:  $C_{n\delta_r}$  of the ATLIT Airplane

Symbol	Description	Reference	Magnitude
$C_{Y\delta_r}$	Change in side force with rudder deflection, per deg	Table 4.5.1.1	0.00299
$l'_v$	Distance parallel to the X-body axis from the c.g. to the quarter-chord of the mean aerodynamic chord of the portion of the vertical tail spanned by the rudder, m (ft)	Figure 2.1.5	4.50 (14.75)
$z'_v$	Distance parallel to the Z-body axis from the c.g. to the quarter chord of the mean aerodynamic chord of the portion of vertical tail spanned by the rudder, m (ft)	Figure 2.1.5	-1.22 (-4.0)
$b_v$	Wing span, m (ft)	Table 2.1.1	12.19 (40.0)

$\alpha$ , deg	$C_{n\delta_r}$ per deg
-4	-0.00104
-2	-0.00105
0	-0.00107
2	-0.00108
4	-0.00108
8	-0.00110
15.9	-0.00110



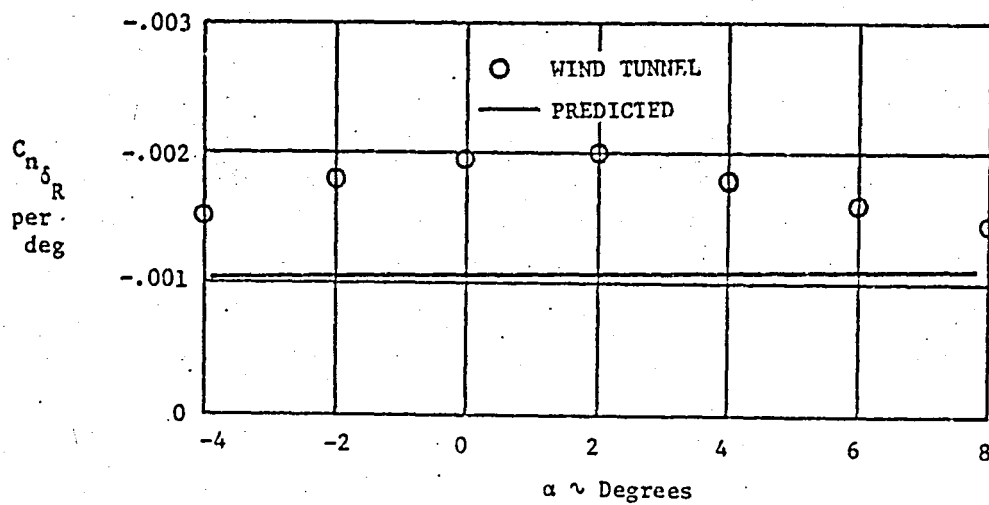


Figure 4.5.3.1: Comparison of predicted  $C_{n\delta_R}$  with wind tunnel data  
 ( $\beta = 0^\circ$ ,  $\delta_R = \pm 10^\circ$ )

## CHAPTER 5

### PREDICTION OF PROPELLER-OFF DYNAMIC DERIVATIVES

In this chapter the propeller-off dynamic (rate) derivatives are calculated. The methods of Reference 3 are used in all cases. The methods are based upon lifting-surface theory and are considered valid for attached flow conditions. It is assumed that the flow will remain attached up to near stall for conventional general aviation airplanes of moderate to high aspect ratio. All computed derivatives are in the stability axis system.

#### 5.1 Roll Damping Derivative, $C_{l_p}$

The roll damping of the airplane is found by considering the contributions of the following components:

- (1) Wing-body
- (2) Horizontal tail
- (3) Vertical tail
- (4) Nacelles

##### 5.1.1 Wing-Body Contribution to $C_{l_p}$

For conventional general aviation airplanes with fuselage-width to wing-span ratios of 0.25 or less, the wing is the dominating factor and the fuselage may be ignored. For zero lift,  $C_{l_p}$  of the wing may be found from Figure 5.1.1.1.

To account for dihedral and non-zero lift conditions, Equation 5.1.1.1 from Reference 3 is used.

$$\left[ (C_{l_p})_{wf} \right]_{M=0} = (C_{l_p})_{w_{C_L=0}} \frac{(C_{L_\alpha})_{w_{C_L=0}}}{(C_{L_\alpha})_{w_{C_L=0}}} \frac{(C_{l_p})_{\Gamma}}{(C_{l_p})_{\Gamma=0}} + (\Delta C_{l_p})_{\text{drag}} \quad (5.1.1.1)$$

where

$(C_{l_p})_{w_{C_L=0}}$  is the zero lift  $C_{l_p}$  obtained from Figure 5.1.1.1

$(C_{L_\alpha})_{w_{C_L}}$  is the propeller off lift curve slope of the wing at the lift coefficient being considered, obtained from Figure 4.1.1.1.

$(C_{L_\alpha})_{w_{C_L=0}}$  is the propeller off lift curve slope of the wing at zero lift

$\frac{(C_{l_p})_{\Gamma}}{(C_{l_p})_{\Gamma=0}}$  is a correction factor which accounts for wing dihedral, obtained from Figure 5.1.1.2

$(\Delta C_{l_p})_{\text{drag}}$  is the drag-induced rolling moment caused by wing drag in roll, obtained from Equation 5.1.1.2 from Reference 3. This increment is small for high aspect ratios.

$$(\Delta C_{l_p})_{\text{drag}} = -\frac{1}{8} \frac{C_{L_w}^2}{\pi A_w \cos^2(\Lambda_{c/4})_w} \left[ 1 + 2 \sin^2 \Lambda_{c/4} \frac{A_w + 2 \cos(\Lambda_{c/4})_w}{A_w + 4 \cos(\Lambda_{c/4})_w} \right] - \frac{1}{8} (C_{D_o})_w \quad (5.1.1.2)$$

where

$(C_{D_o})_w$  is the wing zero lift drag coefficient, obtained from Reference 1.

The low speed wing-body contribution to  $C_{l_p}$  is next modified to account for non-zero Mach number according to Equation 5.1.1.3.

$$\left[ (C_{l_p})_{wf} \right]_M = \frac{A_w + 4 \cos(\Lambda_{c/4})_w}{A_w^2 + 4 \cos(\Lambda_{c/4})_w} \left[ (C_{l_p})_{wf} \right]_{M=0} \quad (5.1.1.3)$$

where

$$B_{2_w} = \sqrt{1 - M^2 \cos^2(\lambda_{c/4})_w}$$

Table 5.1.1.1 summarizes the calculations for the wing-body contribution to  $C_{L_p}$  for the ATLIT airplane.

### 5.1.2 Horizontal-Tail Contribution to $C_{L_p}$

In general the horizontal tail contribution can be found in an analogous manner to the wing contribution. For normal tails it is of the order of one percent on the wing contribution.

For zero dihedral tails in the linear lift region, Equation 5.1.2.1 (from Reference 3) gives the horizontal tail  $C_{L_p}$ , referenced to the wing area. It also accounts for wing interference effects.

$$(C_{L_p})_{hf} = 0.5 \frac{S_h}{S_w} \left( \frac{b_h}{b_w} \right)^2 \frac{\bar{q}_h}{\bar{q}_w} \left[ \frac{l_h + 4 \cos(\lambda_{c/4})_h}{\lambda_h B_{2_h} + 4 \cos(\lambda_{c/4})_h} \right] \left[ (C_{L_p})_{h_{C_L=0}} + (C_{L_p})_{drag} \right] \quad (5.1.2.1)$$

where all quantities are referenced to the horizontal tail and defined in Section 5.1.1.

In Table 5.1.2.1 the horizontal tail calculations are summarized for the ATLIT airplane.

### 5.1.3 Vertical Tail Contribution to $C_{L_p}$

During rolling, the wing induces sidewash on the vertical tail due to unsymmetrical span loading. The equation used to determine the vertical tail contribution, from Reference 3, is

$$(C_{L_p})_v = -57.3 (C'_{L_x})_v \left( \frac{z_v \cos \alpha + l_v \sin \alpha}{b_w} \right) \left( \frac{2(z_v \cos \alpha + l_v \sin \alpha)}{b_w} + \frac{3\sigma}{3 \frac{pb_w}{2v}} \right)$$

(5.1.3.1)

where

$(C_{L_u}^t)_v$  is the vertical tail effective lift curve slope obtained from Section 4.5.1.

$z_v$  is the vertical distance parallel to the Z-body axis from the c.g. to the vertical tail mean aerodynamic chord, obtained from Figure 2.1.3.

$l_v$  is the horizontal distance (parallel to the X-body axis) from the center of gravity to the quarter chord of the mean aerodynamic chord of the vertical tail, from Figure 2.1.3.

$\frac{3\sigma}{pb_w} \frac{1}{2V}$  is the rate of change of sidewash with wing-tip helix angle.

There is no easy way to calculate  $\frac{3\sigma}{pb_w} \frac{1}{2V}$ . This factor varies with angle of attack and is influenced by aspect ratio, sweepback, and tail geometry. A value of .20 was used in Reference 3 and thus was also used here due to lack of better guidelines and the similarity of the airplane in Reference 3 with the ATLIT airplane.

The summary calculations for the vertical tail contribution are given in Table 5.1.3.1 for the ATLIT airplane. It can be seen that the effect of the tail is small.

#### 5.1.4 Nacelle Contribution to $C_{l_p}$

The nacelle contribution to  $C_{l_p}$  is caused by the change in angle of attack induced by the roll rate. Equation 5.1.4.1 from Reference 3 gives this effect.

$$(C_{l_p})_n = -114.6(C_{L_{\dot{\alpha}}})_n \left(\frac{v_T}{b_w}\right)^2 \quad (5.1.4.1)$$

where

$y_T$  is the distance parallel to the Y-axis from the thrust line to the center of gravity

$(C_{L_\alpha})_n$  is the lift curve slope of the nacelles

The value of  $(C_{L_\alpha})_n$  was found by taking the derivative of Equation 5.1.4.2 from Reference 1 (Equation 4.3.5).

$$C_{L_n} = 0.002031 \alpha + 0.0000201 \alpha^2 \quad (5.1.4.2)$$

therefore

$$(C_{L_\alpha})_n = 0.002031 + 0.0000402 \alpha \quad (5.1.4.3)$$

The nacelle contributions are summarized in Table 5.1.4.1.

#### 5.1.5 $C_{L_p}$ of the ATLIT Airplane

The propeller-off  $C_{L_p}$  of the entire airplane is given by Equation 5.1.5.1.

$$(C_{L_p})_{\text{prop off}} = (C_{L_p})_{wf} + (C_{L_p})_{hf} + (C_{L_p})_v + (C_{L_p})_n \quad (5.1.5.1)$$

Table 5.1.5.1 summarizes the propeller-off  $C_{L_p}$  of the ATLIT. Figure 5.1.5.1 presents  $C_{L_p}$  as a function of angle of attack. There is no wind tunnel data available for comparison.

Table 5.1.1.1: Wing-Body Contribution to  $C_{Lp}$

Symbol	Description	Reference	Magnitude
M	Mach number	-	0.031
$B_{2u}$	$\sqrt{1-M^2 \cos^2 \Lambda_{c/4}}$	-	0.997
$A_u$	Wing aspect ratio	Table 2.1.1	10.32
$\Lambda_{c/4}$	Wing quarter-chord sweep angle, deg.	Table 2.1.1	1.835
$\lambda_u$	Wing taper ratio	Table 2.1.1	0.5
$b_u$	Wing span, m (ft)	Table 2.1.1	12.19 (40.0)
$d_u$	Fuselage width at wing, m (ft)	Figure 2.1.4	1.22 (4.0)
$(C_{Lp})_{u, C_L=0}$	Zero lift wing-body $C_{Lp}$ , per rad	Figure 5.1.1.1	-0.525
$z_u'$	Vertical distance from c.g. to quarter-chord of wing root chord, m (ft)	Figure 2.1.4	0.26 (0.36)
$\Gamma$	Wing dihedral angle, deg	Table 2.1.1	7.0
$\frac{(C_{Lp})_{\Gamma}}{(C_{Lp})_{\Gamma=0}}$	Dihedral effect on $C_{Lp}$	Figure 5.1.1.2	1.0
$(C_{Lp})_{u, C_L=0}$	Wing lift curve slope at zero lift, per rad	Figure 4.1.1.1	5.09
$(C_{Lp})_{u, C_L}$	Wing lift curve slope at $C_L$ , per rad	Figure 4.1.1.1	5.09
$(C_D)_u$	Zero lift wing drag	Table 4.12.1.2 Reference 1	0.0097
Summary: $(C_{Lp})_{uf} = -0.5262 - 0.003866 C_{L_u}^2$ per rad			

Table 5.1.2.1: Horizontal Tail Contribution to  $C_{Lp}$

Symbol	Description	Reference	Magnitude
$S_h$	Horizontal tail area, $m^2$ ( $ft^2$ )	Table 2.1.1	3.50 (38.7)
$S_w$	Wing area, $m^2$ ( $ft^2$ )	Table 2.1.1	14.40 (155.0)
$A_h$	Horizontal tail aspect ratio	Table 2.1.1	4.75
$b_h$	Horizontal tail span, m (ft)	Table 2.1.1	4.13 (13.56)
$b_w$	Wing span, m (ft)	Table 2.1.1	12.19 (40.0)
$\frac{q_h}{q_w}$	Dynamic pressure ratio at horizontal tail	Table 4.9.3.3 Reference 1	1.0
$\Lambda_{c/4h}$	Horizontal tail quarter-chord sweep angle, deg	Table 2.1.1	0
$(C_{Lp})_{hC_{L=0}}$	Zero lift $C_{Lp}$ of the horizontal tail	Figure 5.1.1.1	-0.38
$(\Delta C_{Lp})_{h\text{drag}}$	Increment in $C_{Lp}$ due to drag	Equation 5.1.1.2	variable
Summary: $(C_{Lp})_{hf} = -0.005475 - 0.000120 C_{Lh}^2$ , per rad			



Table 5.1.3.1: Vertical Tail Contribution to  $C_{L_p}$

Symbol	Description	Reference	Magnitude
$(C_{L_a}')_v$	Vertical tail effective lift curve slope, per deg	Table 4.5.1.1	0.00507
$z_v$	Vertical distance from the c.g. to the vertical tail mean aerodynamic chord, m (ft)	Figure 2.1.5	-1.22 (-4.0)
$l_v$	Horizontal distance from the c.g. to the quarter-chord of the vertical tail mean aerodynamic chord, m (ft)	Figure 2.1.5	-4.50 (14.75)
$b_w$	Wing span, m (ft)	Table 2.1.1	12.19 (40.0)
$\frac{\partial \sigma}{\partial \frac{pb_w}{2V}}$	Rate of change of sidewash with wingtip helix angle	-	0.20

$\alpha$ , deg	$(C_{L_p})_v$ , per rad (Eq. 5.1.3.1)
-4	-0.00136
-2	-0.00084
0	0.0
2	0.00065
4	0.00112
15.9	-0.00030

Table 5.1.4.1: Nacelle Contribution to  $C_{L_p}$

Symbol	Description	Reference	Magnitude
$(C_{L_p})_n$	Nacelle lift curve slope, per deg	Equation 5.1.4.3	variable
$y_T$	Distance parallel to Y-axis from thrust line to c.g., m (ft)	Figure 2.1.1	1.90 (5.2)
$b_w$	Wing span, m (ft)	Table 2.1.1	12.19 (40.0)

$\alpha$ , deg	$(C_{L_p})_n$ , per deg Eq. 5.1.4.1
-4	-0.00515
-2	-0.00537
0	-0.00559
2	-0.00581
4	-0.00603
15.9	-0.00735

Table 5.1.5.1:  $C_{z_p}$  of the ATLIT Airplane

$\alpha$ , deg	$(C_{z_p})_{wf}$ Table 5.1.1.1	$(C_{z_p})_{hf}$ Table 5.1.2.1	$(C_{z_p})_v$ Table 5.1.3.1	$(C_{z_p})_n$ Table 5.1.4.1	$(C_{z_p})_{prop\ off}$ Eq. 5.1.5.1
-4	-0.5273	-0.0055	-0.00186	-0.00515	-0.5398
-2	-0.5274	-0.0055	-0.00084	-0.00537	-0.5391
0	-0.5277	-0.0055	0.0	-0.00559	-0.5388
2	-0.5282	-0.0055	0.00065	-0.00581	-0.5289
4	-0.5290	-0.0055	0.00112	-0.00603	-0.5394
15.9	-0.5359	-0.0055	-0.00030	-0.00735	-0.5491

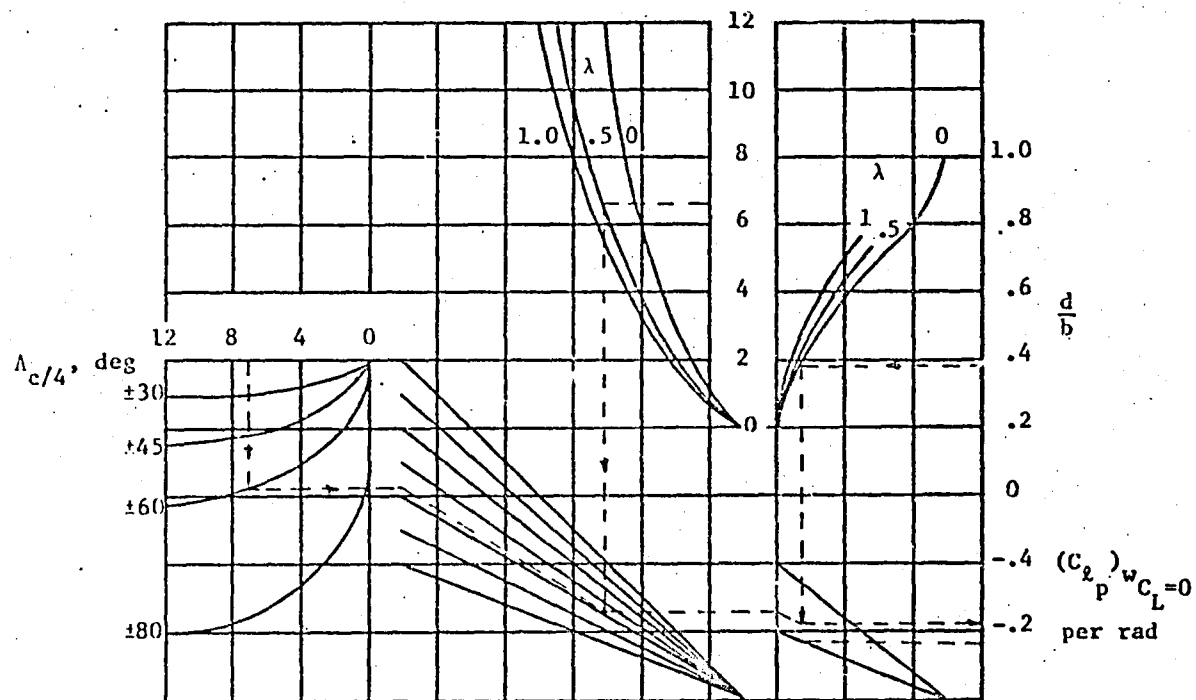


Figure 5.1.1.1: Zero lift wing and horizontal tail  $C_{L_p}$ , including fuselage effects (Reference 3)

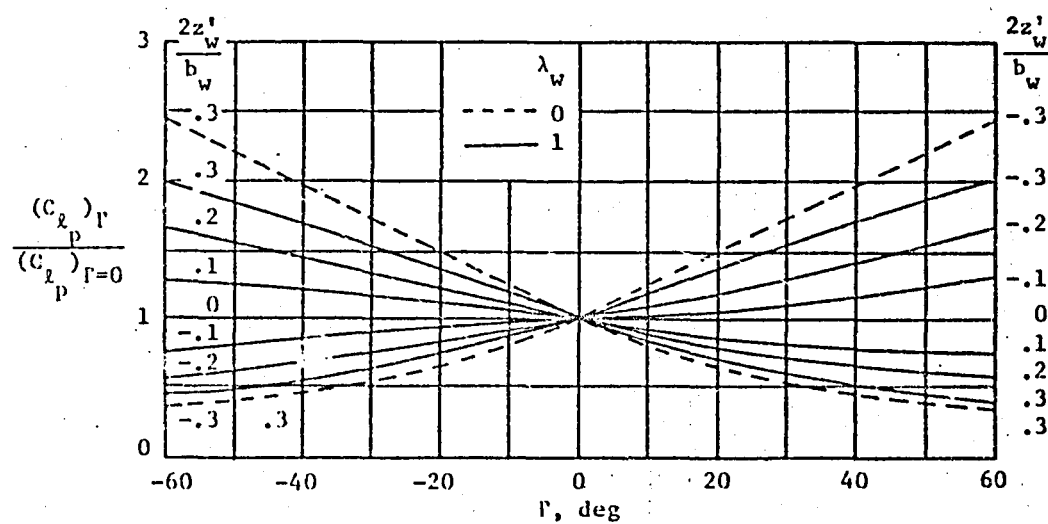


Figure 5.1.1.2: Dihedral contribution to  $C_{l_p}$  (Reference 3)

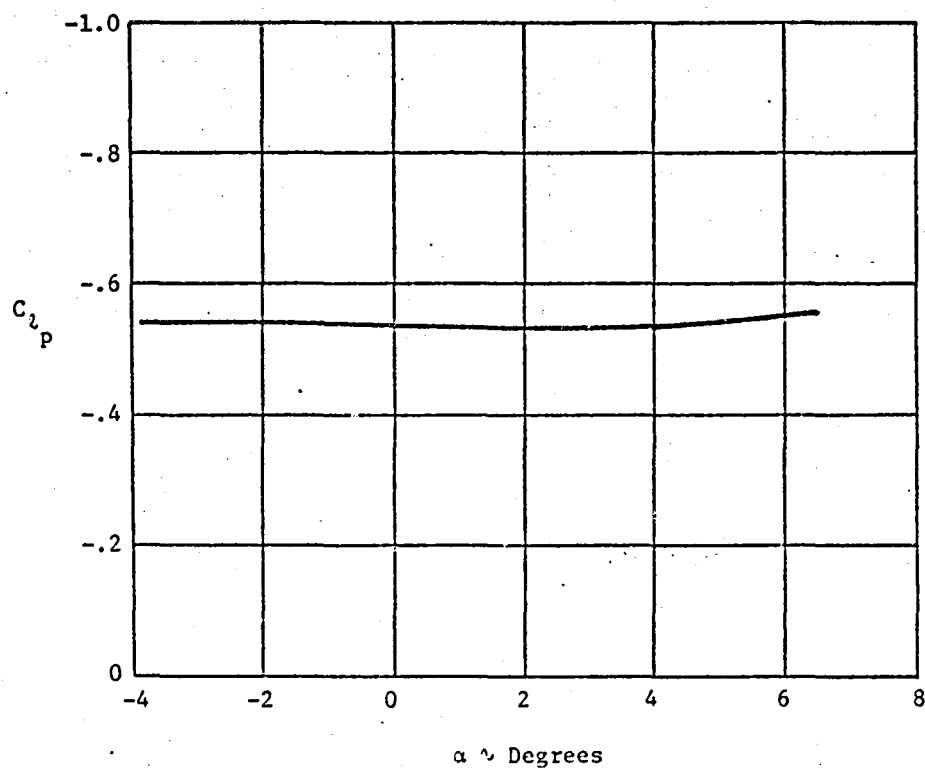


Figure 5.1.5.1: Predicted  $C_{Lp}$  of the ATLIT Airplane  
(propellers removed)

## 5.2 Yaw Damping Derivative, $C_{n_r}$

The yaw damping derivative is largely caused by the vertical tail and to some extent by the wing. The fuselage is of minor importance for conventional general aviation aircraft.

### 5.2.1 Wing Contribution to $C_{n_r}$

A wing undergoing yaw rate will experience asymmetric lift distributions due to changes in velocity. These lift changes cause induced drag changes which produce a yawing moment. Equation 5.2.1.1 from Reference 3 gives the incompressible wing contribution. However, the profile drag term,  $(C_{D_o})_w$ , should be at the appropriate Mach number.

$$(C_{n_r})_w = \left[ \frac{(C_{n_r})_1}{C_{L_w}^2} \right] C_{L_w}^2 + \left[ \frac{(\Delta C_{n_r})_2}{(C_{D_o})_w} \right] (C_{D_o})_w \quad (5.2.1.1)$$

where

$\frac{(C_{n_r})_1}{C_{L_w}^2}$  accounts for the  $C_{n_r}$  due to induced forces resulting from yawing of an isolated wing about its aerodynamic center, obtained from Figure 5.2.1.1.

$\frac{(\Delta C_{n_r})_2}{(C_{D_o})_w}$  accounts for the unsymmetrical spanwise distribution of profile drag, obtained from Figure 5.2.1.2.

If the airplane c.g. does not coincide with the wing aerodynamic center, then the factor  $(C_{n_r})_1/C_{L_w}^2$  must be corrected with the following increment:

$$\left[ \frac{(\Delta C_{n_r})_1}{C_{L_w}^2} \right]_{\bar{x}} = - \left\{ \left( \frac{4 \cos \Lambda_{c/4}}{A_w + 4 \cos \Lambda_{c/4}} + \frac{A_w}{2 \cos \Lambda_{c/4}} \right) \frac{\bar{x}}{\bar{c}_w} \frac{\tan \Lambda_{c/4}}{A_w} + \frac{4 \cos \Lambda_{c/4}}{A_w + 4 \cos \Lambda_{c/4}} \left[ 4 \left( \frac{\bar{x}}{\bar{c}_w} \right)^2 \frac{\tan^2 \Lambda_{c/4}}{A_w^2} \right] \right\} \left[ \frac{(C_{n_r})_1}{C_{L_w}^2} \right]_{\Lambda_{c/4}=0} \quad (5.2.1.2)$$

where

$\bar{x}$  is the longitudinal distance from the c.g. to the wing aerodynamic center.

$\left[ \frac{(C_{n_r})_1}{C_{L_w}^2} \right]_{\Lambda_{c/4}=0}$  is the zero sweep wing damping parameter obtained from Figure 5.2.1.1 for  $\Lambda_{c/4}=0$ .

The general wing contribution equation is now

$$(C_{n_r})_w = C_{L_w}^2 \left\{ \left[ \frac{(C_{n_r})_1}{C_{L_w}^2} \right]_{\bar{x}=0} + \left[ \frac{(C_{n_r})_1}{C_{L_w}^2} \right]_{\bar{x}} \right\} + (C_{n_o})_w \left[ \frac{(\Delta C_{n_r})_2}{(C_{D_o})_w} \right] \quad (5.2.1.3)$$

For the ATLIT the c.g. was assured to be at the wing aerodynamic center.

Table 5.2.1.1 summarizes the wing contribution to  $C_{n_r}$  for the ATLIT.

## 5.2.2 Fuselage Contribution to $C_{n_r}$

For conventional general aviation fuselages the fuselage contribution to  $C_{n_r}$  is small compared to the vertical tail. No analytical design method is presented to calculate  $(C_{n_r})_f$ .

## 5.2.3 Vertical Tail Contribution to $C_{n_r}$

The vertical tail is the primary contributor to  $C_{n_r}$ . Its effect comes from a straight restoring moment due to the sideslip induced by yaw rate and from the lag of wing-fuselage sidewash,  $\frac{\partial \sigma}{\partial \frac{r b_w}{2v}}$ . In



general there is no method to calculate the oscillating sidewash, and in fact Reference 3 suggests that it may not be important. Assuming that the wing cancels the fuselage sidewash, the vertical tail  $C_{n_r}$  is given by Equation 5.2.3.1.

$$(C_{n_r})_v = -114.6(C'_{L_a})_v \left( \frac{l_v \cos \alpha - z_v \sin \alpha}{b_w} \right) \quad (5.2.3.1)$$

where  $(C'_{L_a})_v$ ,  $l_v$ , and  $z_v$  are defined the same as in Section 5.1.3.

Table 5.2.3.1 shows the summary calculations for the vertical tail  $C_{n_r}$  of the ATLIT airplane.

#### 5.2.4 $C_{n_r}$ of the ATLIT Airplane

The propeller-off  $C_{n_r}$  of the ATLIT airplane is given by Equation 5.2.4.1.

$$(C_{n_r})_{\text{prop off}} = (C_{n_r})_w + (C_{n_r})_f + (C_{n_r})_v \quad (5.2.4.1)$$

Table 5.2.4.1 summarizes the propeller-off  $C_{n_r}$  of the ATLIT. Figure 5.2.4.1 shows  $C_{n_r}$  as a function of angle of attack. No wind tunnel data is available for comparison.

Table 5.2.1.1 Wing Contribution to  $C_{n_r}$

Symbol	Description	Reference	Magnitude
$A_w$	Aspect ratio	Table 2.1.1	10.32
$\lambda_w$	Wing taper ratio	Table 2.1.1	0.50
$\Lambda_{c/4}$	Wing quarter-chord sweep angle, deg.	Table 2.1.1	1.835
$\bar{x}$	$\bar{x}_{a.c.} - \bar{x}_{c.g.}$	-	0
$\left[ \frac{(C_{n_r})_1}{C_{L_w}^2} \right]_{\bar{x}=0}$	Wing lift damping parameter	Figure 5.2.1.1	-0.0143
$\left[ \frac{(C_{n_r})_1}{C_{L_w}^2} \right]_{\Lambda_{c/4}=0}$	Zero sweep wing damping parameter	Figure 5.2.1.1	-
$\left[ \frac{(\Delta C_{n_r})_2}{C_{L_w}^2} \right]$	Profile drag damping parameter	Figure 5.2.1.2	-0.289
$C_{L_w}$	Wing lift coefficient	Figure 4.1.1.1	variable
$(C_{D_o})_w$	Wing profile drag coefficient	Table 4.12.1.2 Reference 1	0.00907
Summary: $(C_{n_r})_w = -0.0143 C_{L_w}^2 - 0.0028$ per rad			

Table 5.2.1.1: Vertical Tail Contribution to  $C_{n_r}$

Symbol	Description	Reference	Magnitude
$(C_{L_\alpha})_v$	Vertical tail effective lift curve slope, per deg	Table 4.5.1.1	0.00507
$z_v$	Vertical distance from the c.g. to the vertical tail mean aerodynamic chord, m (ft)	Figure 2.1.5	-1.22 (-4.0)
$l_v$	Horizontal distance from the c.g. to the quarter-chord of the vertical tail mean aerodynamic chord, m (ft)	Figure 2.1.5	4.50 (14.75)
$b_v$	Wing span, m (ft)	Table 2.1.1	12.19 (40.0)

$\alpha$ , deg	$(C_{n_r})_v$ , per rad (Eq. 5.2.3.1)
-4	-0.0757
-2	-0.0774
0	-0.0790
2	-0.0804
4	-0.0816
15.4	-0.0848

Table 5.2.4.1:  $C_{n_r}$  of the ATLIT Airplane

$\alpha, \text{deg}$	$(C_{n_r})_w$ (Table 5.2.1.1)	$(C_{n_r})_v$ Table 5.2.1.2	$(C_{n_r})_{\text{pron off}}$ (Eq. 5.2.4.1)
-4	-0.0028	-0.0757	-0.0735
-2	-0.0030	-0.0774	-0.0804
0	-0.0040	-0.0790	-0.0830
2	-0.0058	-0.0804	-0.0862
4	-0.0084	-0.0816	-0.0900
15.0	-0.0318	-0.0843	-0.1166

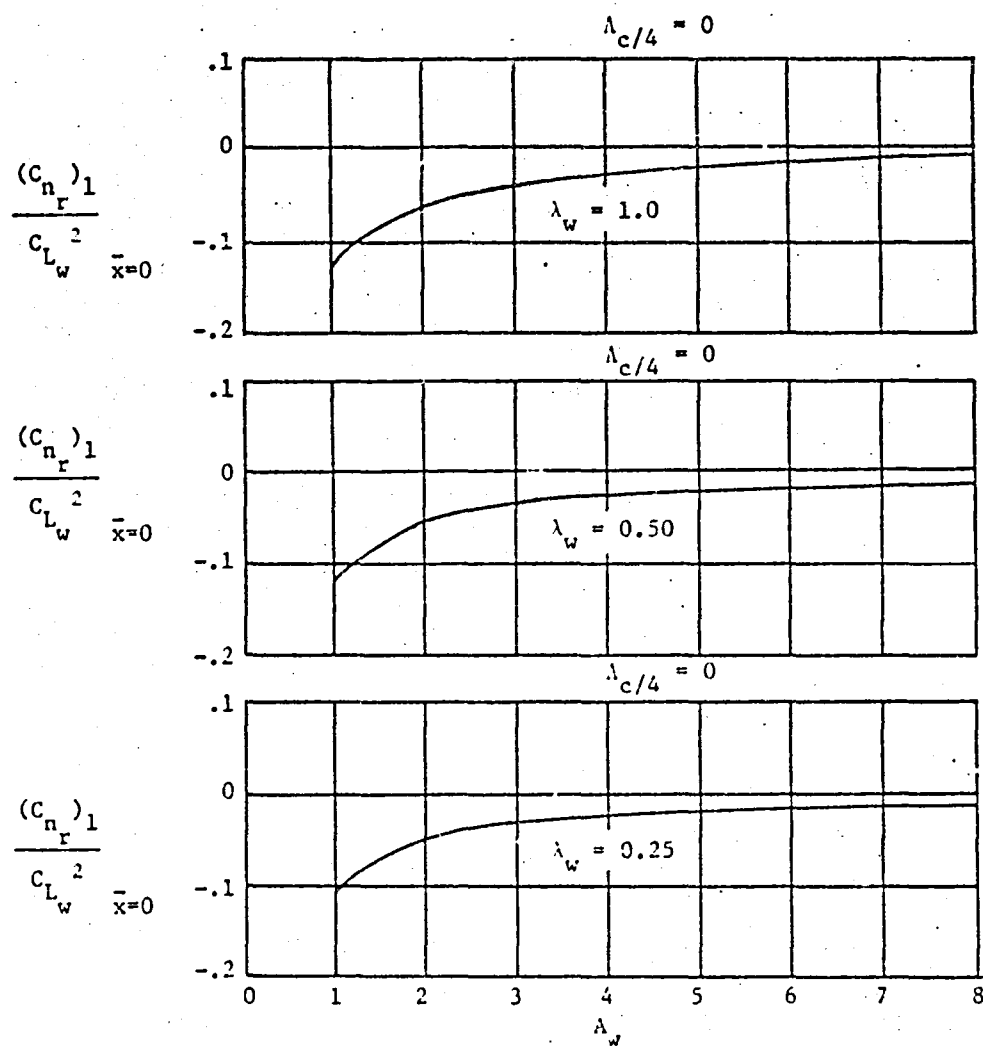


Figure 5.2.1.1: Wing contribution to  $C_{n_r}$  due to lift and induced drag with  $\bar{x} = 0$  (Reference 3)

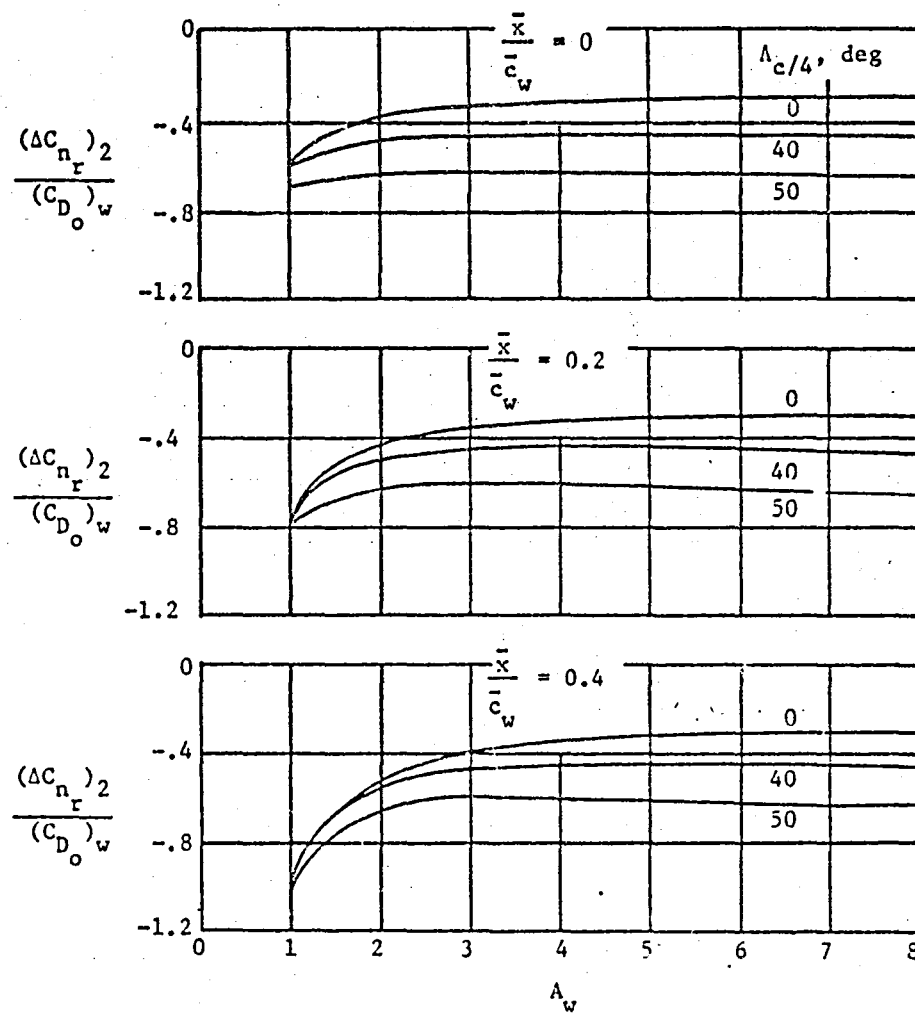


Figure 5.2.1.2: Increment in  $C_{n_r}$  due to wing profile drag  
for  $0.5 < \lambda < 1.0$  (Reference 3)

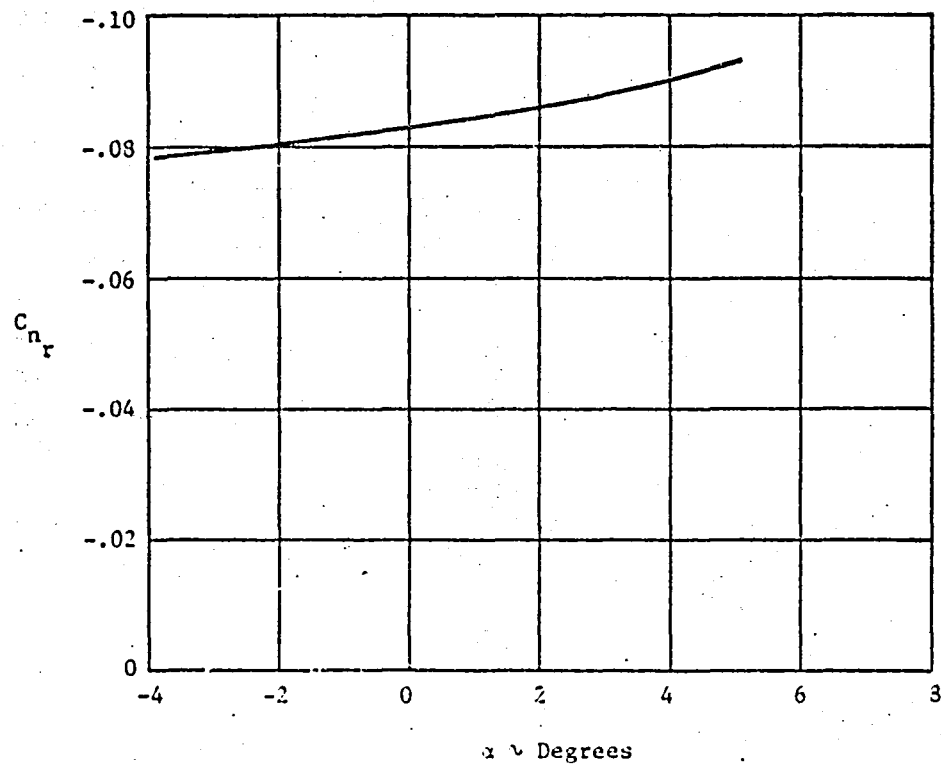


Figure 5.2.4.1: Predicted  $C_{n_r}$  of the ATLIT Airplane  
(propellers removed)

### 5.3 Roll Due to Yaw Rate Derivative, $C_{l_r}$

Rolling moment due to yaw rate is mainly affected by the wing and vertical tail only for airplanes of conventional fuselage shape. Therefore only wing and vertical tail effects are calculated.

#### 5.3.1 Wing Contribution to $C_{l_r}$

The wing contribution to  $C_{l_r}$  is produced by the spanwise lift differential produced by the yaw rate. For unswept wings,  $C_{l_r}$  is linear with lift until  $C_{L_{MAX}}$ . There is also an increment due to geometric dihedral. For zero Mach number, the wing  $C_{l_r}$  is given by Equation 5.3.1.1 from Reference 3.

$$(C_{l_r})_w = \left( \frac{C_{l_r}}{C_{L_w}} \right)_{\Gamma=0} C_{L_w} + \left( \frac{\Delta C_{l_r}}{\Gamma} \right) \frac{\Gamma}{57.3} \quad (5.3.1.1)$$

where

$\left( \frac{C_{l_r}}{C_{L_w}} \right)_{\Gamma=0}$  is the compressible change in  $C_{l_r}$  due to wing lift obtained from Equation 5.3.1.2.

$$\left( \frac{C_{l_r}}{C_{L_w}} \right)_{\Gamma=0} = \frac{1 + \frac{A_w(1-B_2)^2}{2B_2(A_wB_2 + 3 \cos \Lambda_{c/4})} + \left( \frac{A_wB_2 + 2 \cos \Lambda_{c/4}}{A_wB_2 + 4 \cos \Lambda_{c/4}} \right) \left( \frac{\tan^2 \Lambda_{c/4}}{3} \right)}{1 + \frac{A_w + 2 \cos \Lambda_{c/4}}{A_w + 4 \cos \Lambda_{c/4}} \left( \frac{\tan^2 \Lambda_{c/4}}{3} \right)} \frac{C_{l_r}}{C_{L_w}} \quad \Gamma=0 \quad (5.3.1.2)$$

where

$\left( \frac{C_{l_r}}{C_{L_w}} \right)_{\Gamma=M=0}$  is obtained from Figure 5.3.1.1.

$$B_2 = \sqrt{1 - M^2 \cos^2 \Lambda_{c/4}}$$

The dihedral increment is obtained from Equation 5.3.1.3.



$$\frac{\Delta C_{l_r}}{\Gamma} = \frac{1}{12} \left[ \frac{\pi A_w \sin \Lambda_c / 4}{\Lambda_w + 4 \cos \Lambda_c / 4} \right] \quad (5.3.1.3)$$

The valid lift range of Figure 5.3.1.1 is dependent upon sweep angle. For conventional unswept general aviation airplanes, it is usable throughout the linear lift range. Also note that this procedure ignores the effect of center of gravity location. In general there does not appear to be enough data to account for this effect.

Table 5.3.1.1 summarizes the calculations of wing  $C_{l_r}$  for the ATLIT airplane.

### 5.3.2 Vertical Tail Contributions to $C_{l_r}$

Excluding the effects of sidewash,  $\frac{\partial \sigma}{\partial \frac{rb_w}{2v}}$ , as in Section 5.2.3 the aperiodic contribution of the vertical tail to  $C_{l_r}$  can be obtained from Equation 5.3.2.1 from Reference 3. These calculations are summarized in Table 5.3.2.1 for the ATLIT airplane.

$$(C_{l_r})_v = -114.6 (C'_{L_\alpha})_v \left( \frac{z_v \cos \alpha + l_v \sin \alpha}{c_w} \right) \left( \frac{l_v \cos \alpha - z_v \sin \alpha}{b_w} \right) \quad (5.3.2.1)$$

where  $(C'_{L_\alpha})_v$ ,  $z_v$ , and  $l_v$  are defined the same as in Section 5.1.3.

### 5.3.3 $C_{l_r}$ of the ATLIT Airplane

The propeller-off  $C_{l_r}$  of the ATLIT airplane is given by Equation 5.3.3.1.

$$(C_{l_r})_{\text{prop off}} = (C_{l_r})_w + (C_{l_r})_v \quad (5.3.3.1)$$

Table 5.3.3.1 summarizes the propeller-off  $C_{l_r}$  of the ATLIT. Figure 5.3.3.1 shows  $C_{l_r}$  as a function of angle of attack.

Table 5.3.1.1: Wing Contribution to  $C_{L_r}$

Symbol	Description	Reference	Magnitude
M	Mach number	-	0.018
$A_w$	Wing aspect ratio	Table 2.1.1.1	10.32
$\lambda_w$	Wing taper ratio	Table 2.1.1	0.50
$A_{c/4}$	Wing quarter-chord sweep angle, deg	Table 2.1.1	1.835
$\Gamma$	Wing dihedral, deg	Table 2.1.1	7
$C_{L_w}$	Wing lift coefficient	Figure 4.1.1.1	variable
$\left(\frac{C_{L_r}}{C_{L_w}}\right)_{\Gamma=0}$	Wing contribution to $C_{L_r}$ for $M=\Gamma=0$	Figure 5.3.1.1	0.240
$\left(\frac{C_{L_r}}{C_{L_w}}\right)_{\Gamma=0}$	Wing contribution to $C_{L_r}$ for $\Gamma=0$	Equation 5.3.1.2	0.240
$\frac{C_{L_r}}{\Gamma}$	Dihedral contribution to $C_{L_r}$	Equation 5.3.1.3	0.00074
Summary: $(C_{L_r})_w = 0.34 C_{L_w} + 0.00074 \text{ per rad}$			

Table 5.3.2.1: Vertical Tail Contribution to  $C_{L_r}$

Symbol	Description	Reference	Magnitude
$(C_{L_\alpha}')_v$	Vertical tail effective lift curve slope, per deg	Table 4.5.1.1	0.00507
$z_v$	Vertical distance from the c.g. to the vertical tail mean aerodynamic chord, m (ft)	Figure 2.1.5	-1.22 (-4.0)
$l_v$	Horizontal distance from the c.g. to the quarter-chord of the vertical tail mean aerodynamic chord, m (ft)	Figure 2.1.5	4.50 (14.75)
$b_w$	Wing span, m (ft)	Table 2.1.1	12.19 (40.0)

$\alpha$ , deg	$(C_{L_r})_v$ , per rad (Eq. 5.3.2.1)
-4	0.0263
-2	0.0239
0	0.0214
2	0.0138
4	0.0161
15.9	-0.0011

Table 5.3.3.1:  $C_{L_r}$  of the ATLIT Airplane

$\alpha$ , deg	$(C_{L_r})_u$ (Table 5.3.1.1)	$(C_{L_r})_v$ (Table 5.3.2.1)	$(C_{L_r})_{prop\ off}$ (Eq. 5.3.3.1)
-4	-0.0121	0.0253	0.0142
-2	0.0306	0.0239	0.0545
0	0.0732	0.0214	0.0946
2	0.1158	0.0113	0.1346
4	0.1584	0.0161	0.1745
15.9	0.3593	-0.0011	0.3582

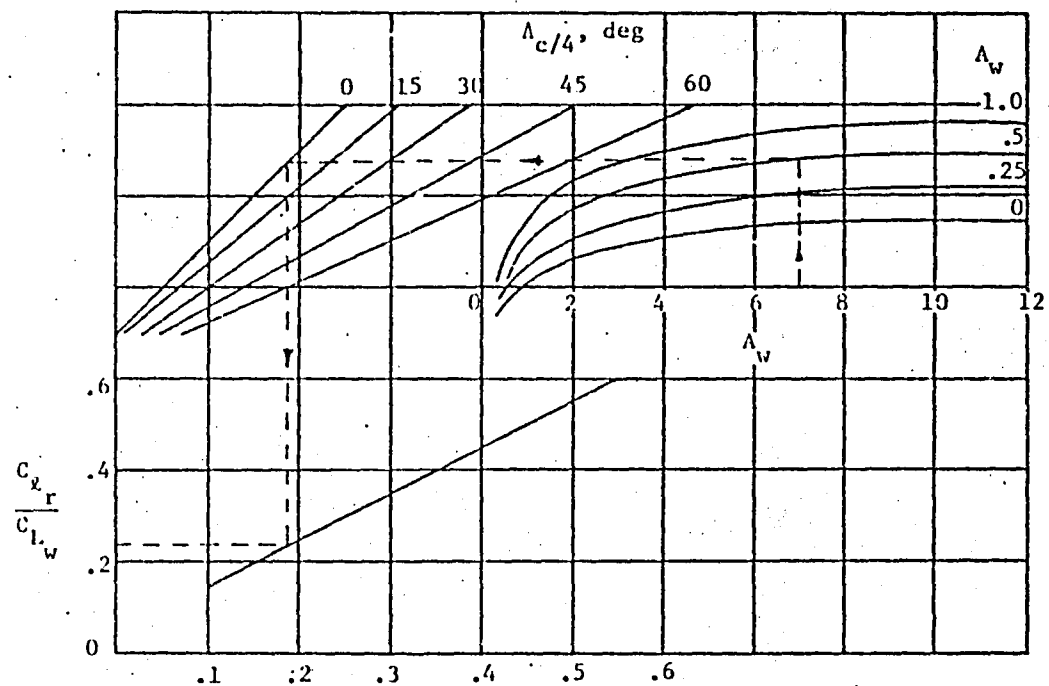


Figure 5.3.1.1: Wing contribution to  $C_{l_r}$  for zero dihedral and incompressible speeds.

(Reference 3)

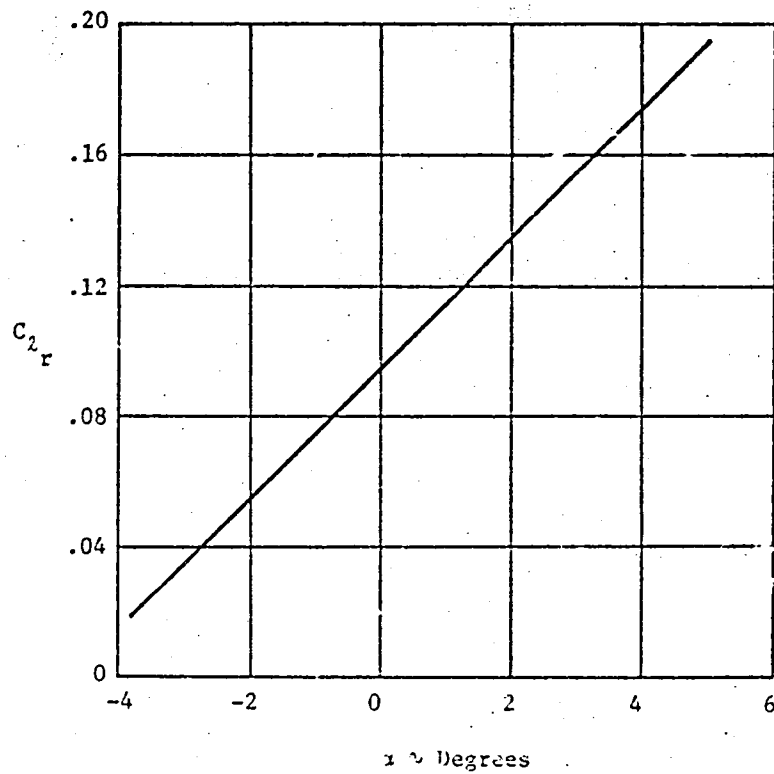


Figure 5.3.3.1: Predicted  $C_{L_r}$  of the ATLIT Airplane  
(propellers removed)

#### 5.4 Yaw Due to Roll Rate Derivative, $C_{n_p}$

In this analysis only the effects of the wing and vertical tail are calculated, since they are the predominating contributors to  $C_{n_p}$ .

##### 5.4.1 Wing Contribution to $C_{n_p}$

Wing rolling produces antisymmetrical lift loading which causes a yawing moment due to induced drag. Additionally roll-induced changes in angle of attack produce a change in viscous drag. Also dihedral effects are considered separate. Equation 5.4.1.1 from Reference 3 gives the wing contribution.

$$(C_{n_p})_w = \left[ \frac{(C_{n_p})_1}{C_{L_w}} \right]_{\Gamma=0} C_{L_w} + \left( \frac{\Delta C_{n_p}}{\Gamma} \right) \frac{\Gamma}{57.3} + \left( \frac{\Delta C_{n_p}}{\partial C_{D_o} / \partial \alpha} \right) \frac{\partial C_{D_o}}{\partial \alpha} \quad (5.4.1.1)$$

where

$\left[ \frac{(C_{n_p})_1}{C_{L_w}} \right]_{\Gamma=0}$  is the zero dihedral contribution of the antisymmetrical lift and induced drag. This term is obtained from Equation 5.4.1.2 from Reference 3 which accounts for tip suction effects and center of gravity location.

$$\frac{C_{n_p1}}{C_{L_w}} = \frac{A_v + 4}{A_v \cos^2 c/4} \left[ 1 + 4 \left( 1 + \frac{\cos^2 c/4}{A_v} \right) \left( \frac{x}{c_v} \frac{\tan^2 c/4}{A_v} + \frac{\tan^2 c/4}{A_v} \right) \right] \left[ \frac{(C_{n_p})_1}{C_{L_w}} \right]_{A_{c/4}=0} + \frac{1}{4 A_v} \left( \tan^2 c/4 + \frac{1}{A_v} \right) - \frac{1}{A_v} \frac{x}{c_v} \quad (5.4.1.2)$$

where

$\left[ \frac{(C_{n_p})_1}{C_{L_w}} \right]_{A_{c/4}=0}$  is found from Figure 5.4.1.2

To correct for compressibility effects, Equation 5.4.1.3 is used.

$$\left[ \frac{(C_{n_p})_1}{C_{L_w}} \right]_{\Gamma=0} = \left( \frac{A_w + 4 \cos \Lambda_{c/4}}{A_w B_2 + 4 \cos \Lambda_{c/4}} \right) \left( \frac{A_w B_2 + \frac{1}{2} (A_w B_2 + \cos \Lambda_{c/4}) \tan^2 \Lambda_{c/4}}{A_w + \frac{1}{2} (A_w + \cos \Lambda_{c/4}) \tan^2 \Lambda_{c/4}} \right) \left[ \frac{(C_{n_p})_1}{C_{L_w}} \right]_{\Gamma=M=0} \quad (5.4.1.3)$$

where

$$B_2 = \sqrt{1 - M^2 \cos^2 \Lambda_{c/4}}$$

To account for dihedral, Equation 5.4.1.4 from Reference 3 is used.

$$\left( \frac{\Delta C_{n_p}}{\Gamma} \right) = - \left( \frac{\tan \Lambda_{c/4}}{4} + \frac{3}{A_w} \frac{\bar{x}}{\bar{c}_w} \right) (C_{L_p})_{\Gamma=0} \quad (5.4.1.4)$$

where  $(C_{L_p})_{\Gamma=0}$  is from Section 5.1.1 with compressibility effects included.

The viscous drag parameter  $\frac{\Delta C_{n_p}}{\partial C_D^o / \partial \alpha}$  in Equation 5.4.1.1 is found from Figure 5.4.1.2. The second viscous drag term  $\frac{\partial C_D^o}{\partial \alpha}$  is found by plotting viscous drag versus angle of attack, as shown in Figure 5.4.1.3, and graphically determining the slopes. An analytical method is presented in Appendix A. Reference 3 points out the importance of including the viscous drag contribution to  $C_{n_p}$  in obtaining good correlation with wind-tunnel results.

Table 5.4.1.1 summarizes the wing contribution to  $C_{n_p}$ .

#### 5.4.2 Vertical Tail Contribution to $C_{n_p}$

The vertical tail contribution is given by Equation 5.4.2.1 from Reference 3, where all terms are as defined in Section 5.1.3.



$$(C_{n_p})_v = -57.3(C_{L_{\alpha}})_v \left( \frac{l_v \cos \alpha + z_v \sin \alpha}{b_w} \right) \left[ \frac{2(z_v \cos \alpha + l_v \sin \alpha)}{b_w} + \frac{\partial \sigma}{\partial \frac{pb_w}{2V}} \right] \quad (5.4.2.1)$$

Table 5.4.2.1 summarizes the vertical tail contribution to  $C_{n_p}$  for the ATLIT airplane.

#### 5.4.3 $C_{n_p}$ of the ATLIT Airplane

Considering the wing and vertical tail as the only significant contributors,  $C_{n_p}$  is given in Equation 5.4.3.1.

$$(C_{n_p})_{\text{prop off}} = (C_{n_p})_w + (C_{n_p})_v$$

The calculations are summarized in Table 5.4.3.1 and plotted as a function of angle of attack in Figure 5.4.3.1. No wind tunnel data is available for comparison.

Table 5.4.1.1: Wing Contribution to  $C_{np}$

Symbol	Description	Reference	Magnitude
M	Mach number	-	0.018
$B_2$	$1-M^2 \cos^2 \Lambda_{c/4}$	-	0.997
$A_w$	Wing aspect ratio	Table 2.1.1	10.32
$t_w$	Wing taper ratio	Table 2.1.1	0.50
$\Lambda_{c/4}$	Wing quarter-chord sweep angle, deg	Table 2.1.1	1.835
$b_w$	Wing span, ft	Table 2.1.1	12.19 (40.0)
$\bar{c}_w$	Wing mean aerodynamic chord, ft	Table 2.1.1	1.225 (4.018)
$\bar{x}$	$\bar{x}_{a.c.} - \bar{x}_{c.g.}$	-	0
$\frac{(C_{n_p})_1}{C_{L_w}} \lambda=0$	Zero sweep damping parameter	Figure 5.4.1.1	-0.372
$\frac{(C_{n_p})_2}{C_{L_w}} \Gamma=0$	Zero dihedral damping parameter	Equation 5.4.1.3	-0.0752
$\frac{\Delta C_{n_p}}{\Delta \Gamma}$	Dihedral contribution to $C_{n_p}$	Equation 5.4.1.4	0.0042
$\frac{\Delta C_{n_p}}{\Delta C_D}$	Viscous drag parameter	Figure 5.4.1.2	2.059
$\frac{\Delta C_{n_p}}{\Delta C_D}$	Viscous drag rate of change	Figure 5.4.1.3	variable

$\lambda$ , deg	$(C_{n_p})_1$ , per rad (Eq. 5.4.1.1)
-4	0.50452
-2	-0.0334
0	-0.32219
2	-0.33155
4	-0.34091
11.3	-0.34531

Table 5.4.2.1: Vertical Tail Contribution to  $C_{n_p}$

Symbol	Description	Reference	Magnitude
$(C_{L_\alpha})_v$	Vertical tail effective lift curve slope, per deg	Table 4.5.1.1	0.00507
$z_v$	Vertical distance from the c.g. to the vertical tail mean aerodynamic chord, m (ft)	Figure 2.1.5	-1.22 (-4.0)
$l_v$	Horizontal distance from the c.g. to the quarter-chord of the mean aerodynamic chord of the vertical tail, m (ft)	Figure 2.1.5	4.50 (14.75)
$b_w$	Wing span, m (ft)	Table 2.1.1	12.19 (40.0)
$\frac{\partial \alpha}{\partial \delta_w}$	Rate of change of sidewash with wing tip helix angle	-	0.20

$\alpha$ , deg	$(C_{n_p})_v$ , per rad (Eq. 5.4.2.1)
-4	0.00198
-2	0.00272
0	0
2	-0.00280
4	-0.00507
15.9	-0.02333

Table 5.4.3.1:  $C_{n_p}$  of the ATLIT Airplane

$\alpha$ , deg	$(C_{n_p})_w$ (Table 5.4.1.1)	$(C_{n_p})_v$ (Table 5.4.2.1)	$(C_{n_p})_{prop off}$
-4	0.00452	0.00198	0.00650
-2	-0.00884	0.00272	-0.00612
0	-0.02219	0	-0.02219
2	-0.03355	-0.00280	-0.03635
4	-0.04690	-0.00567	-0.05257
15.9	-0.09534	-0.02333	-0.11867

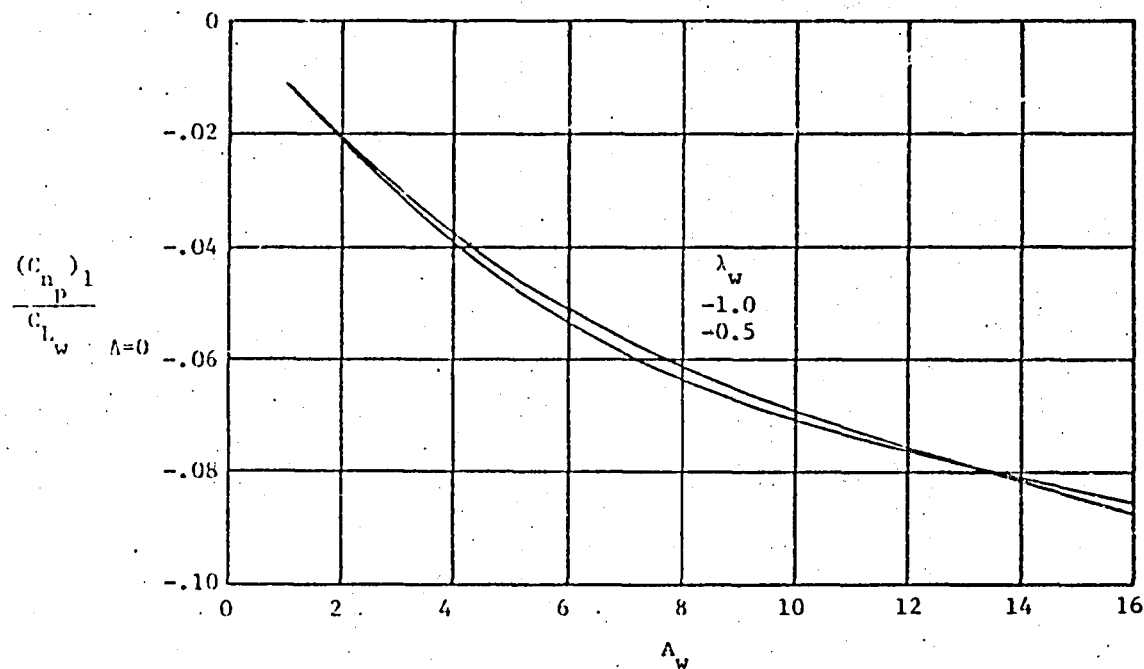


Figure 5.4.1.1: Wing contribution to  $C_{np}$  for unswept wings due to antisymmetrical lift and induced drag (Reference 3)

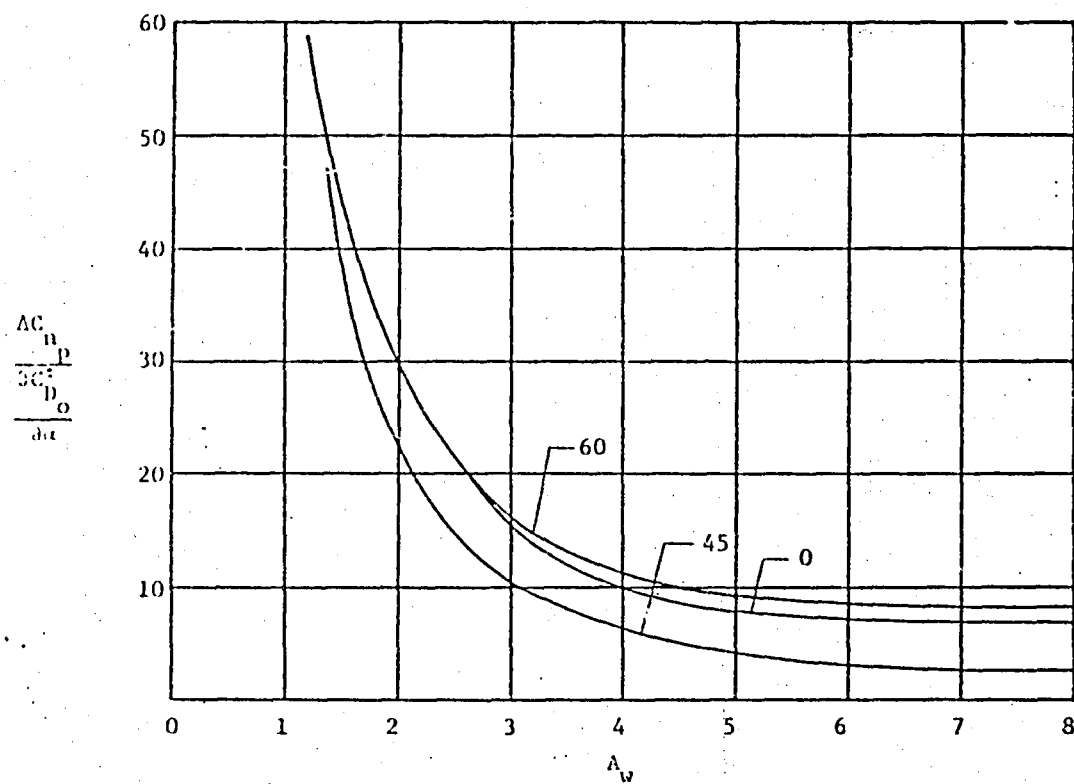


Figure 5.4.1.2: Increment in  $C_{np}$  due to wing viscous drag (Reference 3)

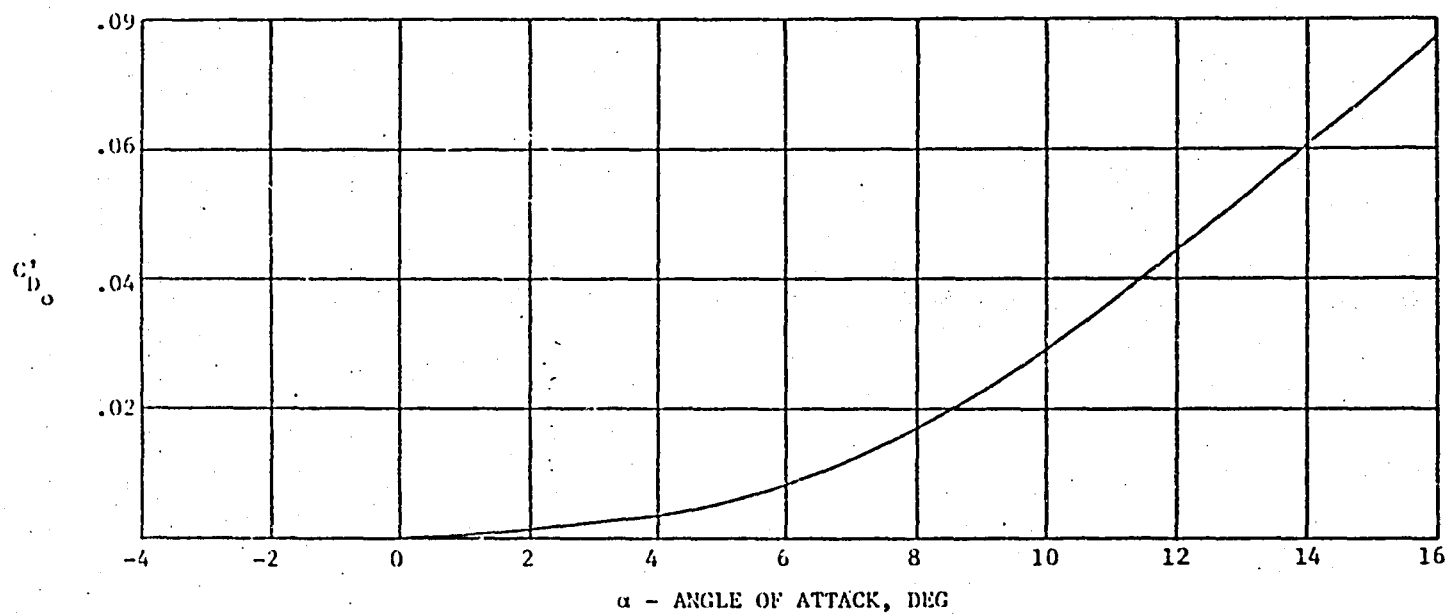


Figure 5.4.1.3: Variation of viscous drag with angle of attack for ATLIT (Reference 3)

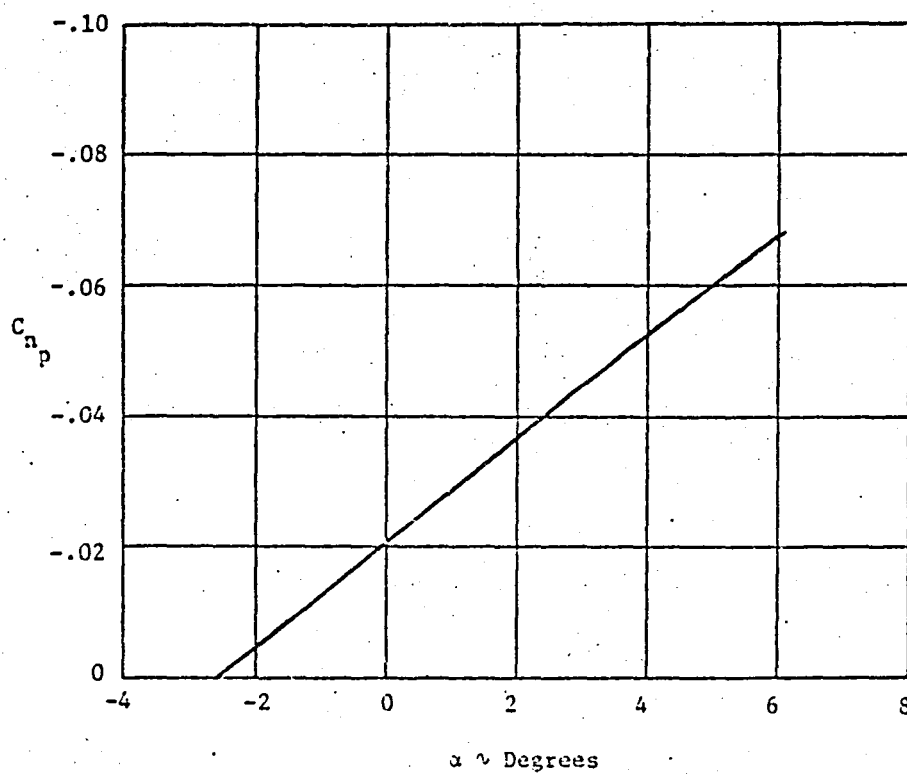


Figure 5.4.3.1: Predicted  $C_{np}$  of the ATLIT Airplane  
(propellers removed)



## CHAPTER 6

### PREDICTION OF POWER-ON AERODYNAMIC CHARACTERISTICS

The effects of power especially from a propeller operation can have a significant effect on the aerodynamic characteristics of an airplane. Due to the highly interactive nature of the propeller slipstream, there are usually a number of effects that must be considered. Power effects are considered at three power settings:

$T'_c=0$ ,  $T'_c=.0195$ ,  $T'_c=.1970$ , where  $T'_c = \text{Thrust} / \bar{q}_\infty S_w$ .

#### 6.1 Propeller Power Effects on Static Stability Characteristics

For conventional twin engine airplanes the vertical tail is assumed to be outside the propeller slipstream. Therefore the propellers are assumed to produce no sidewash or dynamic pressure change at the vertical tail.

##### 6.1.1 Power Effects on $C_{Y_3}$

The contribution of power to  $C_{Y_3}$  is given in Equation 6.1.1.1.

$$(\Delta C_{Y_3})_{\text{power}} = (\Delta C_{Y_3})_{N_p} + (\Delta C_{Y_3})_{n(\Delta \bar{q})} + (\Delta C_{Y_3})_{n(\sigma_p)} \quad (6.1.1.1)$$

where

$(\Delta C_{Y_3})_{N_p}$  is the normal side force caused by the propellers

$(\Delta C_{Y_3})_{n(\Delta \bar{q})}$  is the change in nacelle  $C_{Y_3}$  due to the increased dynamic pressure

$(\Delta C_{Y_3})_{n(\sigma_p)}$  is the change in nacelle  $C_{Y_3}$  due to power induced sidewash at the nacelle.

The normal force contribution  $(\Delta C_{Y_\beta})_{N_p}$  is given by Equation 6.1.1.2 from Reference 3.

$$(\Delta C_{Y_\beta})_{N_p} = -\frac{nf}{57.3} (C_{N_\alpha})_p \left( \frac{S_p/\text{prop}}{S_w} \right) \quad (6.1.1.2)$$

where

$n$  is number of propellers

$f$  is the propeller inflow factor, obtained from Figure 6.1.1.1

$S_p/\text{prop}$  is the disc area of each propeller

$(C_{N_\alpha})_p$  is the normal force parameter at  $T'_c=0$  given by Equation 6.1.1.3 from Reference 3.

$$(C_{N_\alpha})_p = (C_{N_\alpha})_{p, K_N=80.7} \left[ 1 + .8 \left( \frac{K_N}{80.7} \right) - 1 \right] \quad (6.1.1.3)$$

where

$K_N$  is the empirical normal force factor of the propeller given by

$$K_N = 262 \left( \frac{b}{R_p} \right) 0.3R_p + 262 \left( \frac{b}{R_p} \right) 0.6R_p + 135 \left( \frac{b}{R_p} \right) 0.9R_p \quad (6.1.1.4)$$

where  $\left( \frac{b}{R_p} \right)$  is the ratio of blade width to propeller radius at the given propeller radius stations.

$[(C_{N_\alpha})_{p, K_N=80.7}]$  is the normal coefficient at  $K_N=80.7$  obtained from Figure 6.1.1.2 as a function of blade angle  $\beta'$ .

The change in  $C_{Y_\beta}$  of the nacelles due to increased dynamic pressure is given by Equation 6.1.1.5 from Reference 3.

$$(\Delta C_{Y_\beta})_{n(\Delta \bar{q})} = [(C_{Y_\beta})_n]_{\text{prop off}} \frac{\Delta \bar{q}_n}{\bar{q}_w} \quad (6.1.1.5)$$

where

$[(\Delta C_{Y_B})_n]_{\text{prop off}}$  is nacelle contribution with propellers off as determined in Section 4.1.3

$\frac{\Delta \bar{q}_n}{\bar{q}_\infty}$  is the increase in dynamic pressure at the nacelle due to power given by

$$\frac{\Delta \bar{q}_n}{\bar{q}_\infty} = \frac{S_w (T'_c / \text{prop})}{\pi R_p^2} \quad (6.1.1.6)$$

The effect of propeller induced sidewash on the nacelle contribution is given by Equation 6.1.1.7 from Reference 3.

$$(\Delta C_{Y_B})_{n(\sigma_p)} = -[(\Delta C_{Y_B})_n]_{\text{prop off}} \left( \frac{\partial \sigma_p}{\partial \beta} \right) \left( 1 + \frac{\Delta \bar{q}_n}{\bar{q}_\infty} \right) \quad (6.1.1.7)$$

where

$\frac{\partial \sigma_p}{\partial \beta}$  is the change in propeller induced sidewash with sideslip given by

$$\frac{\partial \sigma_p}{\partial \beta} = C_1 + C_2 (C_{N_a})_p \quad (6.1.1.8)$$

where  $C_1$  and  $C_2$  are obtained from Figure 6.1.1.3.

Table 6.1.1.1 (a) - (c) summarizes the effects of power on  $C_{Y_B}$  of the ATLIT airplane. The power on  $C_{Y_B}$  is given by

$$C_{Y_B} = (C_{Y_B})_{\text{prop off}} + (\Delta C_{Y_B})_{\text{power}} \quad (6.1.1.9)$$

Table 6.1.1.1 (d) gives the power on  $C_{Y_B}$  of the ATLIT. Figure 6.1.1.4 presents this data as a function of angle of attack and power setting,  $T'_c$ . Also the calculated values are compared with wind-tunnel results.

### 6.1.2 Power Effects on $C_{n_3}$

The effect of power on  $C_{n_3}$  is due to power-induced changes in  $C_{Y_3}$ . The same factors are assumed in the power increment.

$$(\Delta C_{n_3})_{\text{power}} = (\Delta C_{n_3})_{N_p} + (\Delta C_{n_3})_{n(\Delta \bar{q})} + (\Delta C_{n_3})_{n(\sigma_p)} \quad (6.1.2.1)$$

The propeller normal side force increment,  $(\Delta C_{n_3})_{N_p}$ , is given by Equation 6.1.2.2.

$$(\Delta C_{n_3})_{N_p} = (\Delta C_{Y_3})_{N_p} \left( \frac{x_p \cos \alpha + z_p \sin \alpha}{b_w} \right) \quad (6.1.2.2)$$

where

$(\Delta C_{n_3})_{N_p}$  is the increment in  $C_{Y_3}$  due to propeller normal force, from Section 6.1.1

$x_p$  is the longitudinal distance from the propeller to the airplane c.g.

$z_p$  is the vertical distance from the thrust line to the airplane c.g.

$b_w$  is the wing span.

Since the two nacelle increments,  $(\Delta C_{n_3})_{n(\Delta \bar{q})}$  and  $(\Delta C_{n_3})_{n(\sigma_p)}$ , act through the same moment arm, their effect can be considered together.

$$(\Delta C_{n_3})_{n(\Delta \bar{q})} + (\Delta C_{n_3})_{n(\sigma_p)} = [(\Delta C_{Y_3})_{n(\Delta \bar{q})} + (\Delta C_{Y_3})_{n(\sigma_p)}] \left( \frac{x_n \cos \alpha + z_n \sin \alpha}{b_w} \right) \quad (6.1.2.3)$$

where

$(\Delta C_{Y_3})_{n(\Delta \bar{q})}$  and  $(\Delta C_{Y_3})_{n(\sigma_p)}$  are the power increments to  $C_{Y_3}$  due to the nacelles, obtained from Section 6.1.1

$x_n$  is the longitudinal distance from the nacelle center of pressure to the airplane c.g., from Figure 2.1.4

$z_n$  is the vertical distance from the nacelle center of pressure to the airplane c.g., from Figure 2.1.4.

The total airplane  $C_{n_3}$  is

$$(C_{n_3}) = (C_{n_3})_{\text{prop off}} + (\Delta C_{n_3})_{\text{power}} \quad (6.1.2.4)$$

Table 6.1.2.1 presents the summary calculations for the ATLIT airplane.

Figure 6.1.2.1 gives total  $C_{n_3}$  as a function of angle of attack and power setting, comparing it with wind tunnel results.

### 6.1.3 Power Effects on $C_{l_3}$

Two power effects on  $C_{l_3}$  are considered. The first is caused by the normal force produced by the propeller. The second is the result of the lateral shift of the portion of the wing immersed in the propeller slipstream. Equation 6.1.3.1 gives the total power increment.

$$(\Delta C_{l_3})_{\text{power}} = (\Delta C_{l_3})_{N_p} + (\Delta C_{l_3})(\Delta \bar{q} + \epsilon_p) \quad (6.1.3.1)$$

where

$(\Delta C_{l_3})_{N_p}$  is the propeller normal side force increment given by Equation 6.1.3.2.

$$(\Delta C_{l_3})_{N_p} = (\Delta C_{Y_3})_{N_p} \left( \frac{-z_p \cos \alpha + x_p \sin \alpha}{b_w} \right) \quad (6.1.3.2)$$

where all terms are the same as in Equation 6.1.2.2.

When the airplane is yawed, the center of pressure of the wing shifts proportional to  $\tan(3-\tau_p)$ . On one wing the shift will be inboard, while on the opposite wing the shift will be outboard. The net result is a rolling moment. Equation 6.1.3.3, from Reference 3, gives the increment.

$$(\Delta C_{L_B})_{(\Delta q + \epsilon_p)} = \left(\frac{1}{2}\right) \frac{1}{57.3} \frac{x'_p}{b_w} [(\Delta C_L)_w(\Delta q) + (\Delta C_L)_w(\epsilon_p)] \left(1 - \frac{\partial \sigma_p}{\partial \beta}\right) \quad (6.1.3.3)$$

where

the factor 1/2 accounts for fuselage interference effects

$x'_p$  is the distance from the propeller to the wing quarter-chord along the thrust line

$b_w$  is the wing span

$(\Delta C_L)_w(\Delta q)$  is the increase in lift due to power-induced increased dynamic pressure, obtained from Table 5.1.3.2, of Reference 1

$(\Delta C_L)_w(\epsilon_p)$  is the increment in lift due to power-induced downwash, obtained from Table 5.1.3.2 of Reference 1

$\frac{\partial \sigma_p}{\partial \beta}$  is the power-induced sidewash from Equation 6.1.1.3.

Table 6.1.2.1 presents the summary calculations for the ATLIT airplane. Total airplane  $C_{L_B}$  is:

$$C_{L_B} = (C_{L_B})_{\text{prop off}} + (\Delta C_{L_B})_{\text{power}} \quad (6.1.3.4)$$

Figure 6.1.3.1 gives total  $C_{L_B}$  as a function of angle of attack and power setting and compares it with wind-tunnel results.

#### 6.1.4 Rudder Control

As previously mentioned, it is assumed that the vertical tail is outside the propeller slipstream and is not affected by power settings. As a result, there is no predicted change in  $C_{L_R}$ ,  $C_{n_R}$ , or  $C_{\dot{\delta}_R}$  with power. Figure 6.1.4.1 shows the full-scale wind-tunnel results for the power effect on rudder control derivatives. As shown, it appears that there is very little power interaction.

### 6.1.5 Spoiler Control

For an airplane with ailerons, the effect of power on roll control should be small. For spoilers it depends on the proximity of the spoilers to the propeller slipstream. For the ATLIT airplane, the portion of the spoiler immersed in the slipstream is quite small at low angle of attack, and none of the spoiler is immersed at higher angles of attack. (For data on the immersed span as a function of power and angle of attack, see Table 5.1.3.2 of Reference 1.) For this reason it was assumed that the effect of power would be small. In addition there is no wind tunnel data that gives the isolated effect of power. The spoiler derivatives were therefore assumed not to vary with power setting.

Table 6.1.1.1(a): Increment of  $C_{Y_3}$  Due to Propeller Normal Force,  $(\Delta C_{Y_3})_{N_p}$

Symbol	Description	Reference	Magnitude
$n$	Number of propellers	-	2
$R_p$	Propeller radius, m (ft)	Table 2.1	0.97 (3.17)
$b_p$	Width of propeller blade, m (ft)	-	0.127 (0.417) at $.3 R_p$
			0.157 (0.515) at $.5 R_p$
			0.108 (0.354) at $.7 R_p$
$S_{p/prop}$	Propeller disc area, $\pi R_p^2$ , m <sup>2</sup> (ft <sup>2</sup> )	-	2.93 (31.57)
$\frac{S_{p/prop}}{8 R_p^2}$	Propeller correlation parameter	-	variable
$f$	Propeller inflow parameter	Figure 6.1.1.1	variable
$K_n$	Propeller side force parameter	Equation 6.1.1.4	
$\beta'$	Propeller blade angle, deg	-	variable
$(C_{N_3})_{K_N=40.7}$	Propeller normal force parameter	Figure 6.1.1.2	variable
$(C_{N_1})_p$	Propeller normal force derivative	Equation 6.1.1.3	variable

$T_C$	$(C_{N_1})_p$ (Eq. 6.1.1.3)	$(\Delta C_{Y_3})_{N_p}$ (Eq. 6.1.1.2)
0	0.10	-0.0007
0.0315	0.10	-0.0008
0.1375	0.10	-0.0024



Table 6.1.1.1(b): Change in Nacelle  $C_{Y_3}$  Due to Increased Dynamic Pressure,  $(\Delta C_{Y_3})_n(\Delta \bar{q})$

Symbol	Description	Reference	Magnitude
$(\Delta C_{Y_3})_n$ prop off	Propeller off nacelle contribution to $C_{Y_3}$ , per deg	Table 4.1.1.1	-0.000475
$\frac{\Delta \bar{q}_n}{\bar{q}_3}$	Power induced change in nacelle dynamic pressure	Equation 6.1.1.6	variable

$T_c$	$\frac{\Delta \bar{q}_n}{\bar{q}_3}$ (Eq. 6.1.1.6)	$(\Delta C_{Y_3})_n(\Delta \bar{q})$ , per deg (Eq. 6.1.1.5)
0	0	0
0.0915	0.2249	-0.000107
0.1970	0.4336	-0.000230

Table 6.1.1.1(c): Change in Nacelle  $C_{Y_3}$  Due to Propeller Sidewash,  $(\Delta C_{Y_3})_n(\sigma_p)$

Symbol	Description	Reference	Magnitude
$(\Delta C_{Y_3})_n$ prop off	Propeller off nacelle contribution to $C_{Y_3}$ , per deg	Table 4.1.3.1	-0.000476
$\frac{\partial \sigma_p}{\partial \delta}$	Change in propeller sidewash with sideslip	Equation 6.1.1.8	variable
$C_1, C_2$	Propeller sidewash factors	Figure 6.1.1.3	variable
$\frac{\Delta q_n}{q_\infty}$	Power induced change in nacelle dynamic pressure	Equation 6.1.1.6	variable

$T'_C$	$\frac{\partial \sigma_p}{\partial \delta}$ (Eq. 6.1.1.5)	$\frac{\Delta q_n}{q_\infty}$ (Eq. 6.1.1.6)	$(\Delta C_{Y_3})_n(\sigma_p)$ , per deg (Eq. 6.1.1.7)
0	0.025	0	0.000012
0.0915	0.1033	0.2249	0.000060
0.1970	0.1554	0.4836	0.000110

Table 6.1.1.1(d): Power-on  $C_{Y_3}$  of the ATLIT Airplane

$T_c$	$(C_{Y_3})_{prop}$ off (Table 4.1.5.1)	$(\Delta C_{Y_3})_{N_p}$ (Table 6.1.1.1(a))	$(\Delta C_{Y_3})_{n(\Delta \bar{q})}$ (Table 6.1.1.1(b))	$(\Delta C_{Y_3})_{n(\sigma_p)}$ (Table 6.1.1.1(c))	$(C_{Y_3})_{power}$ on, per d. (Eq. 6.1.1)
0	-0.00977	-0.0007	0	0.000012	-0.01046
0.0915	-0.00977	-0.0008	-0.000107	0.000060	-0.01062
0.1970	-0.00977	-0.0008	-0.000230	0.000110	-0.01069

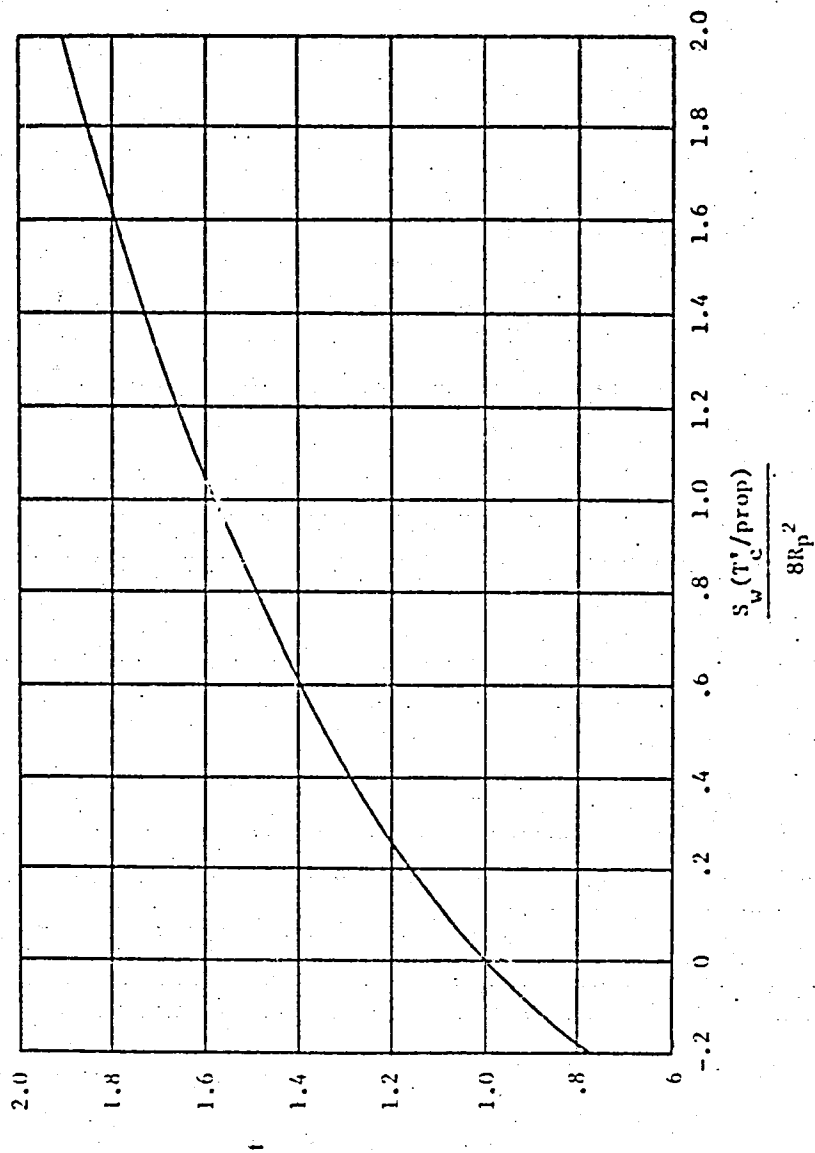


Figure 6.1.1.1: Propeller inflow factor (Reference 3)

$[(C_{N\alpha})_P]_{K_N=80.7}$

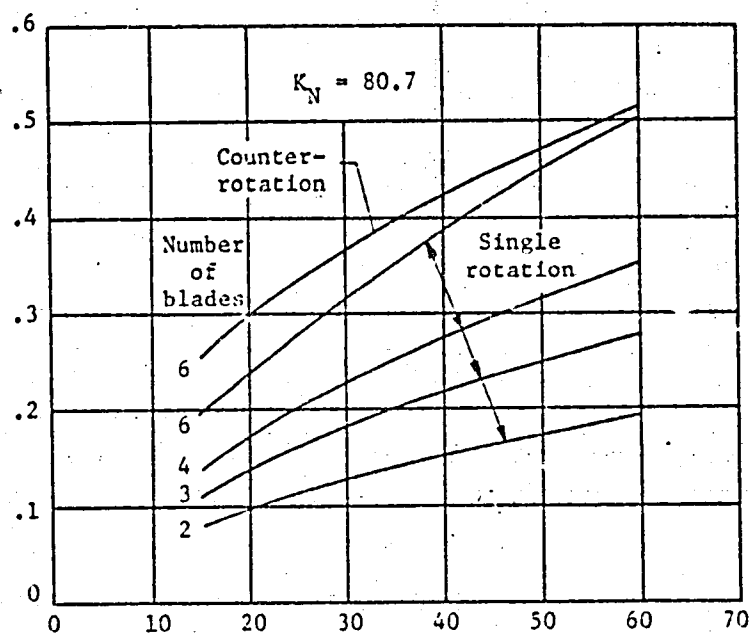


Figure 6.1.1.2: Propeller normal force parameter (Reference 3)

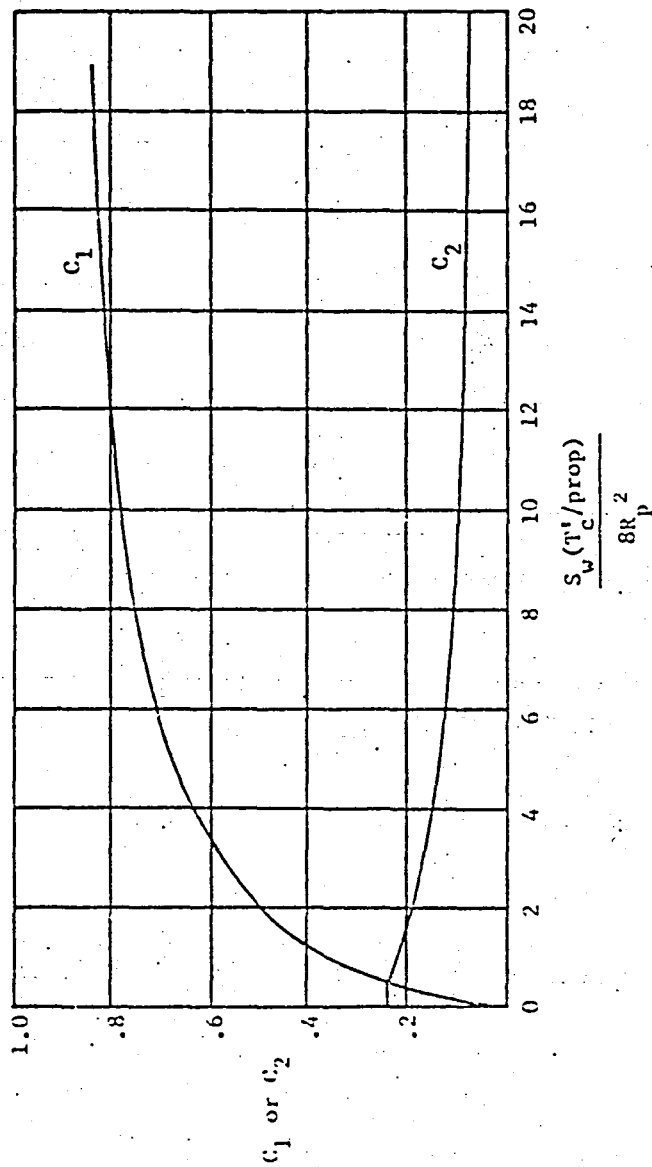


Figure 6.1.1.3: Factors for determining propeller downwash (Reference 3)

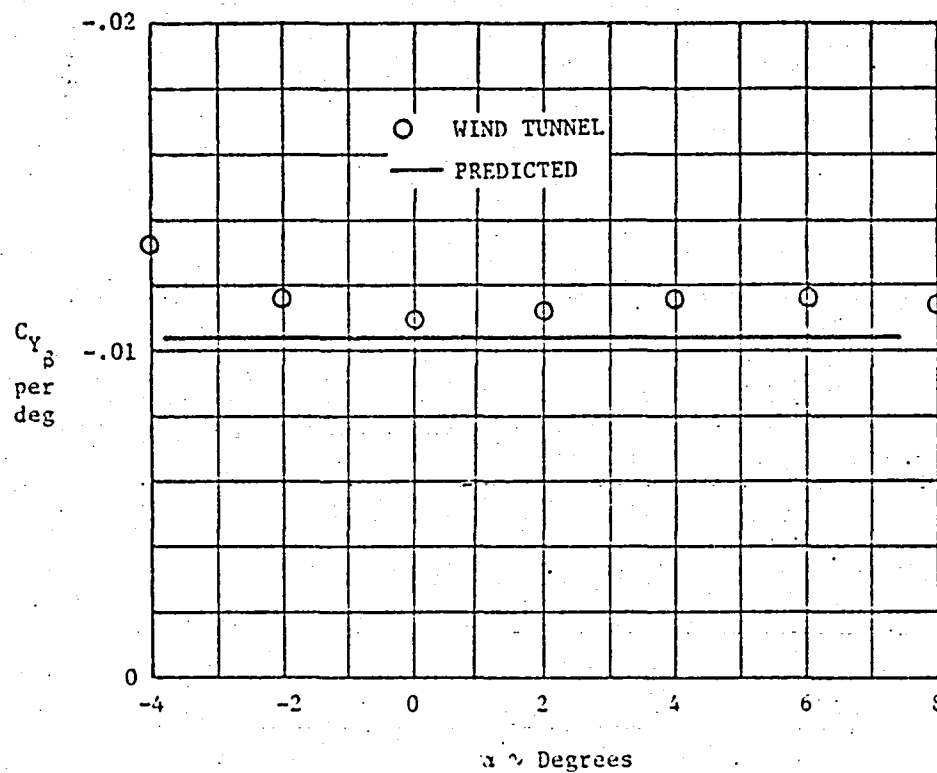


Figure 6.1.1.4: Comparison of predicted power-on  $C_{L_2}$  with wind tunnel data ( $T'_C = .1970$ )

Table 6.1.2.1: Power-on  $C_{n_3}$  of the ATLIT Airplane

Symbol	Description	Reference	Magnitude
$(\Delta C_{Y_B})_{N_P}$	Propeller normal side force derivative, per deg	Table 6.1.1.1	variable
$x_p$	Longitudinal distance from the propeller to the c.g., m (ft)	Figure 2.1.4	2.10 (6.59)
$z_p$	Vertical distance from the propeller to the c.g., m (ft)	Figure 2.1.4	-0.127 (-0.417)
$b_w$	Wing span, m (ft)	Table 2.1.1	12.19 (40.0)
$(\Delta C_{Y_B})_{n(\Delta \bar{q})}$	Change in nacelle $C_{Y_B}$ due to increased dynamic pressure, per deg	Table 6.1.1.1	variable
$(\Delta C_{Y_B})_{n(\sigma_p)}$	Change in nacelle $C_{Y_B}$ due to propeller sidewash, per deg	Table 6.1.1.1	variable
$x_n$	Longitudinal distance from nacelle center of pressure to airplane c.g., m (ft)	Figure 2.1.4	0.97 (3.17)
$z_n$	Vertical distance from nacelle center of pressure to airplane c.g., m (ft)	Figure 2.1.4	-0.15 (-0.50)

$\alpha$ , deg	$(C_{n_3})_{prop off}$	$(\Delta C_{n_3})_{N_P}$			$(\Delta C_{n_3})_{n(\Delta \bar{q})} + (\Delta C_{n_3})_{n(\sigma_p)}$			$(C_{n_3})_{power on}$		
		$T_c'$			$T_c'$			$T_c'$		
		0	0.0915	0.1970	0	0.0915	0.1970	0	0.0915	0.1970
-4	0.00189	-0.0001	-0.0001	-0.0001	0	0	-0.00001	0.00179	0.00179	0.00178
-2	0.00190	-0.0001	-0.0001	-0.0001	0	0	-0.00001	0.00180	0.00180	0.00179
0	0.00191	-0.0001	-0.0001	-0.0001	0	0	-0.00001	0.00181	0.00181	0.00180
2	0.00195	-0.0001	-0.0001	-0.0001	0	0	-0.00001	0.00185	0.00185	0.00184
4	0.00194	-0.0001	-0.0001	-0.0001	0	0	-0.00001	0.00184	0.00184	0.00183



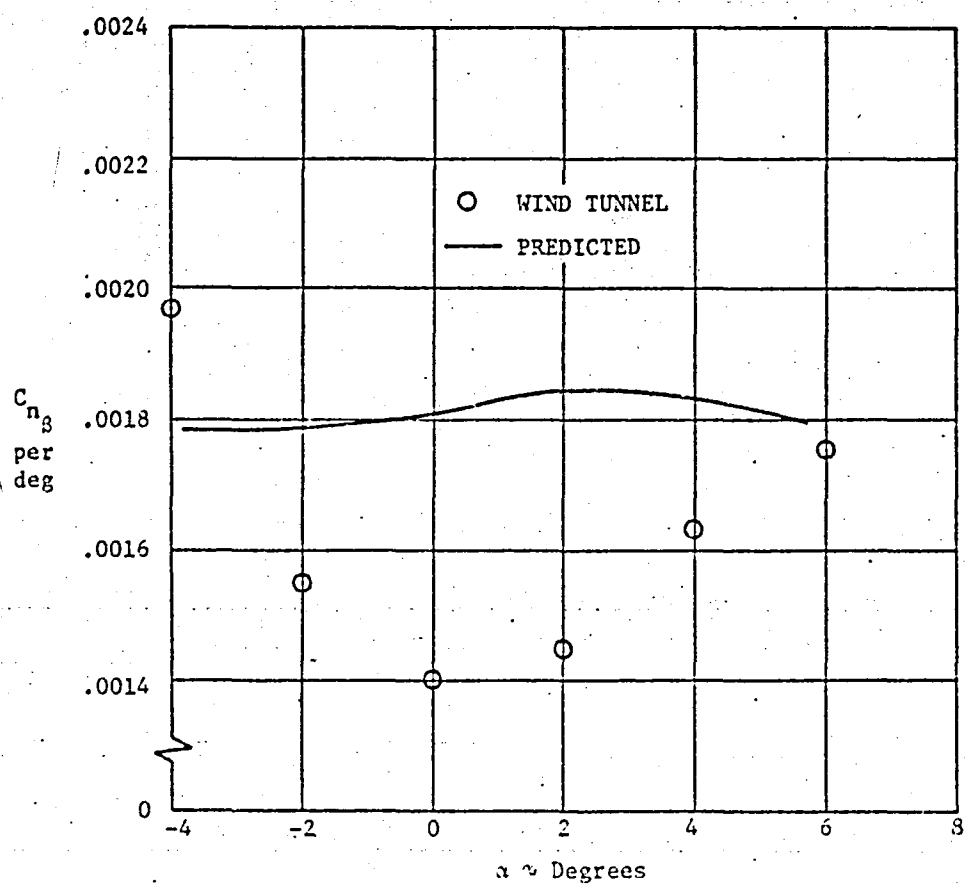


Figure 6.1.2.4: Comparison of predicted power-on  $C_{n_3}$  with wind tunnel data on the ATLIT Airplane ( $T'_c = .1970$ )

Table 6.1.3.1: Power on  $C_{L_3}$  of the ATLIT Airplane

Symbol	Description	Reference	Magnitude
$(\Delta C_{Y_3})_{N_p}$	Propeller normal side force derivative, per deg.	Table 6.1.1.1	variable
$z_p$	Longitudinal distance from the propeller to the c.g., m (ft)	Figure 2.1.4	2.10 (6.59)
$x_p$	Vertical distance from the propeller to the c.g., m (ft)	Figure 2.1.4	-0.127 (-0.417)
$b_w$	Wing span, m (ft)	Table 2.1.1	12.19 (40.0)
$x'_p$	Distance along thrust line from the propeller to the wing quarter-chord, m (ft)	Figure 2.1.4	
$(\Delta C_L)_{u(\Delta \bar{q})}$	Increase in lift due to power-induced increased dynamic pressure	Table 5.1.3.2 Reference 1	variable
$(\Delta C_L)_{u(\epsilon_p)}$	Increment in lift due to power-induced downwash	Table 5.1.3.2 Reference 1	variable
$\frac{\partial C_D}{\partial \beta}$	Power-induced sidewash derivative	Equation 6.1.1.8	variable

$\alpha$ , deg	$(C_{L_3})_{\text{prop off}}$	$(\Delta C_{L_3})_{N_p}$			$(\Delta C_{L_3})_{(\Delta \bar{q} + \epsilon_p)}$			$(C_{L_3})_{\text{power on}}$		
		$T'_c$			$T'_c$			$T'_c$		
		0	0.0915	0.1970	0	0.0915	0.1970	0	0.0915	0.1970
-4	-0.00212	0	0	0	0	0.0001	0.0002	-0.00212	-0.00202	-0.00192
-2	-0.00209	0	0	0	0	0.0002	0.0004	-0.00209	-0.00189	-0.00169
0	-0.00296	-0.00001	-0.00001	-0.00001	0	0.0003	0.0005	-0.00207	-0.00177	-0.00157
2	-0.00203	-0.00001	-0.00001	-0.00001	0	0.0003	0.0007	-0.00204	-0.00174	-0.00134
4	-0.00200	-0.00002	-0.00002	-0.00002	0	0.0004	0.0008	-0.00202	-0.00162	-0.00122
5.9	-0.00174	-0.00004	-0.00004	-0.00004	0	0.0004	0.0009	-0.00178	-0.00138	-0.00088

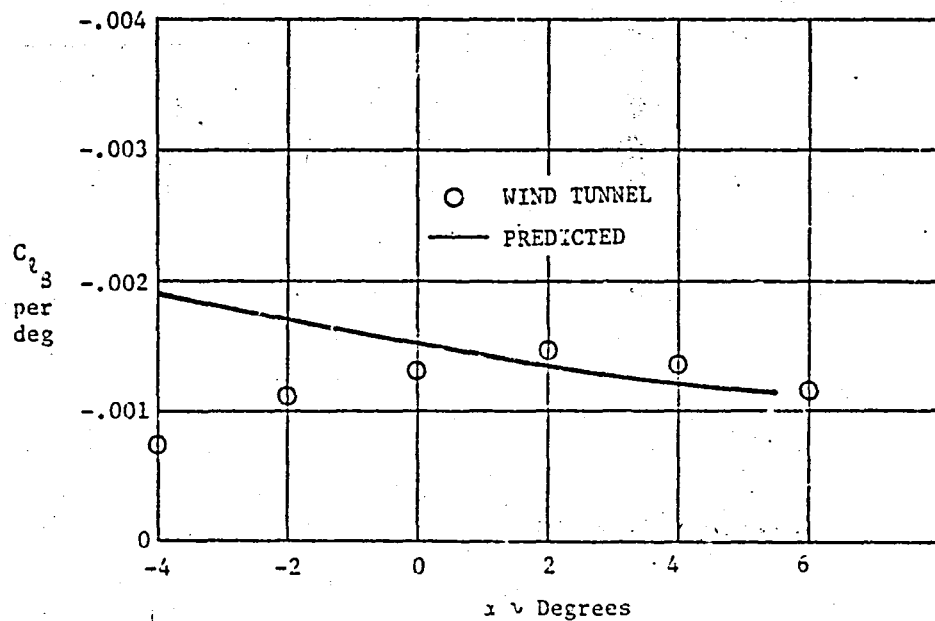


Figure 6.1.3.1: Comparison of predicted power-on  $C_{L3}$  with wind tunnel data on the ATLIT Airplane ( $T'_C = .1970$ )

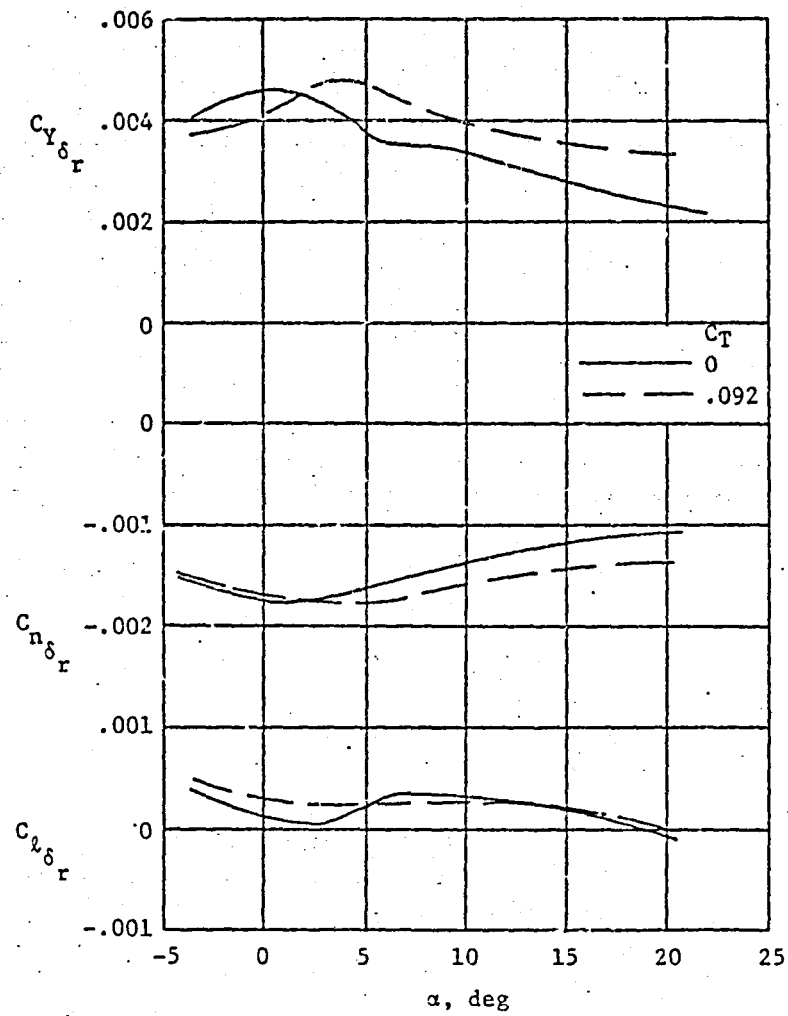


Figure 6.1.4.1: Effect of power on rudder derivatives from full scale wind-tunnel tests (Reference 2)

## 6.2 Propeller Power Effects on Dynamic Derivatives

### 6.2.1 Power Effects on $C_{l_p}$

The power increment on  $C_{l_p}$  is caused by the propeller normal force, the change in dynamic pressure over the immersed sections of the airplane and the change in downwash in the slipstream. The power-induced normal force contribution is given in Equation 6.2.1.1.

$$(\Delta C_{l_p})_{N_p} = -114.6 (C_{L_\alpha})_{N_p} \left( \frac{y_T}{b_w} \right)^2 \quad (6.2.1.1)$$

where

$(C_{L_\alpha})_{N_p}$  is the lift curve slope of the propeller normal force obtained from Figure 6.2.1.1 (from Table 5.1.3.1 of Reference 1)

$y_T$  is the lateral distance from the thrust line to the airplane center of gravity

$b_w$  is the wing span.

The increment in wing  $C_{l_p}$  due to power is given by Equation 6.2.1.2 from Reference 3.

$$\begin{aligned} (\Delta C_{l_p})_{w(\Delta \bar{q} + \epsilon_p)} &= (\Delta C_{l_p})_{w(\Delta \bar{q})} + (\Delta C_{l_p})_{w(\epsilon_p)} \\ &= -114.6 [(\Delta C_{L_\alpha})_{w(\Delta \bar{q})} + (\Delta C_{L_\alpha})_{w(\epsilon_p)}] \left( \frac{y_T}{b_w} \right)^2 \end{aligned} \quad (6.2.1.2)$$

where

$(\Delta C_{L_\alpha})_{w(\Delta \bar{q})}$  is the change in lift curve slope due to the power-induced change in dynamic pressure obtained from Figure 6.2.1.2 (from Table 5.1.3.2 of Reference 1)

$(\Delta C_{L_\alpha})_{w(\epsilon_p)}$  is the change in lift curve slope due to the power-induced change in downwash obtained from Figure 6.2.1.3 (from Table 5.1.3.2 of Reference 1).

The change in nacelle  $C_{L_p}$  with power is given by Equation 6.2.1.3 from Reference 3.

$$\begin{aligned} (\Delta C_{L_p})_{n(\Delta \bar{q} + \epsilon_p)} &= (\Delta C_{L_p})_{n(\Delta \bar{q})} + (\Delta C_{L_p})_{n(\epsilon_p)} \\ &= (C_{L_p})_{n_{prop}} \left[ \frac{\Delta \bar{q}}{\bar{q}_\infty} - \frac{\left( \frac{\partial \epsilon_p}{\partial \alpha_p} \right)}{\left( 1 - \frac{\partial \epsilon_u}{\partial \alpha_w} \right)} \left( 1 + \frac{\Delta \bar{q}}{\bar{q}_\infty} \right) \right] \end{aligned} \quad (6.2.1.3)$$

where

$(C_{L_p})_{n_{prop}}$  is the propeller off nacelle contribution from Section 5.1.4.

$\frac{\Delta \bar{q}}{\bar{q}_\infty}$  is the increment in dynamic pressure behind the propeller obtained from Table 5.1.3.2 of Reference 1.

$\frac{\partial \epsilon_p}{\partial \alpha_p}$  is the rate of change of propeller downwash with propeller angle of attack, obtained from Table 5.1.3.2 of Reference 1.

$\frac{\partial \epsilon_u}{\partial \alpha}$  is the propeller upwash gradient obtained from Table 5.1.3.1 of Reference 1.

Total airplane  $C_{L_p}$  is

$$C_{L_p} = (C_{L_p})_{prop} + (\Delta C_{L_p})_{w(q+\epsilon_p)} + (\Delta C_{L_p})_{n(\Delta \bar{q} + \epsilon_p)} \quad (6.2.1.4)$$

Table 6.2.1.1 summarizes the power effects on the ATLIT airplane.

Figure 6.2.1.4 shows total  $C_{L_p}$  as a function of angle of attack and power setting. There is no wind-tunnel data available for comparison.

#### 6.2.2 Power Effects on $C_{n_r}$

No analytical methods of computing the power increment to  $C_{n_r}$  were found. In Reference 3, the  $C_{n_r}$  variation with power was deduced by comparing with wind-tunnel data on a similar model. The same percentage change in  $C_{n_r}$  with power for the model was assumed.

This procedure is highly empirical and cannot readily be extended to other aircraft. A comparison of the prediction of Reference 3 with flight test data shows poor correlation. In view of this and due to the fact that no experimental ATLIT data is yet available, the power effect on  $C_{n_r}$  is not predicted for the ATLIT.

#### 6.2.3 Power Effects on $C_{l_r}$

Reference 3 shows that the effect of power on  $C_{l_r}$  is due mainly to wing contributions and that even this effect is negligible (of the order of 1%). For this reason the power effects of  $C_{l_r}$  are not computed for the ATLIT and are assumed zero.

#### 6.2.4 Power effects on $C_{n_p}$

Due to a lack of design data and for the same reasons as those outlined above, the power effect on  $C_{n_p}$  for the ATLIT airplane is assumed zero.

Table 6.2.1.1: Power-on  $C_{L_p}$  of the ATLIT Airplane

Symbol	Description	Reference	Magnitude
$(C_{L_a})_{N_p}$	Lift curve slope of the propeller normal force	Figure 6.2.1.1	variable
$y_T$	Lateral distance from thrust line to airplane c.g., m (ft)	Figure 2.1.1	1.39 (6.21)
$b_w$	Wing span, m (ft)	Table 2.1.1	12.19 (40.0)
$(\Delta C_{L_a})_{w(\Delta \bar{q})}$	Change in wing lift curve slope due to power-induced change in dynamic pressure	Figure 6.2.1.2	variable
$(\Delta C_{L_a})_{w(c_p)}$	Change in wing lift curve slope due to propeller downwash	Figure 6.2.1.3	variable
$\frac{\Delta \bar{q}}{\bar{q}_\infty}$	Change in dynamic pressure behind the propeller	Table 5.1.3.2 Reference 1	variable
$(C_{L_p})_{n_{prop off}}$	Propeller off $C_{L_p}$ of the nacelles	Table 5.1.4.1	variable
$\frac{\partial c_n}{\partial \alpha_p}$	Rate of change of propeller downwash with propeller angle of attack	Table 5.1.3.2 Reference 1	
$\frac{\partial c_u}{\partial \alpha}$	Propeller upwash gradient	Table 5.1.3.1 Reference 1	-.155



a, deg	$(C_L)_p$ prop off (Table 5.1.5.1)	$(\Delta C_L)_p N_p$ (Eq. 6.2.1.1)			$(\Delta C_L)_p w(\Delta q + \epsilon_p)$ (Eq. 6.2.1.2)		
		$T'_c$			$T'_c$		
		0	.0915	.1970	0	.0915	.1970
-4	-0.5398	-0.00099	-0.00110	-0.00121	0.00083	-0.0153	-0.0367
-2	-0.5391	-0.00099	-0.00110	-0.00121	0.00083	-0.0152	-0.0340
0	-0.5388	-0.00099	-0.00110	-0.00121	0.00083	-0.0152	-0.0284
2	-0.5289	-0.00099	-0.00110	-0.00121	0.00083	-0.0152	-0.0284
4	-0.5394	-0.00099	-0.00110	-0.00121	0.00083	-0.0105	-0.0284
15.0	-0.5491	-0.00099	-0.00110	-0.00121	-0.00276	0	0.0028

$(\Delta C_L)_p n(\Delta q + \epsilon_p)$ (Eq. 6.2.1.3)			$(C_L)_p$ , per rad (Eq. 6.2.1.4)		
$T'_c$			$T'_c$		
0	.0915	.1970	0	.0915	.0970
0.00011	-0.00059	-0.00146	-0.5399	-0.5567	-0.5792
0.00012	-0.00062	-0.00153	-0.5391	-0.5560	-0.5758
0.00012	-0.00064	-0.00159	-0.5388	-0.5557	-0.5700
0.00013	-0.00067	-0.00165	-0.5289	-0.5459	-0.5602
0.00013	-0.00069	-0.00171	-0.5394	-0.5517	-0.5707
0.00016	-0.00085	-0.00209	-0.5527	-0.5511	-0.5496

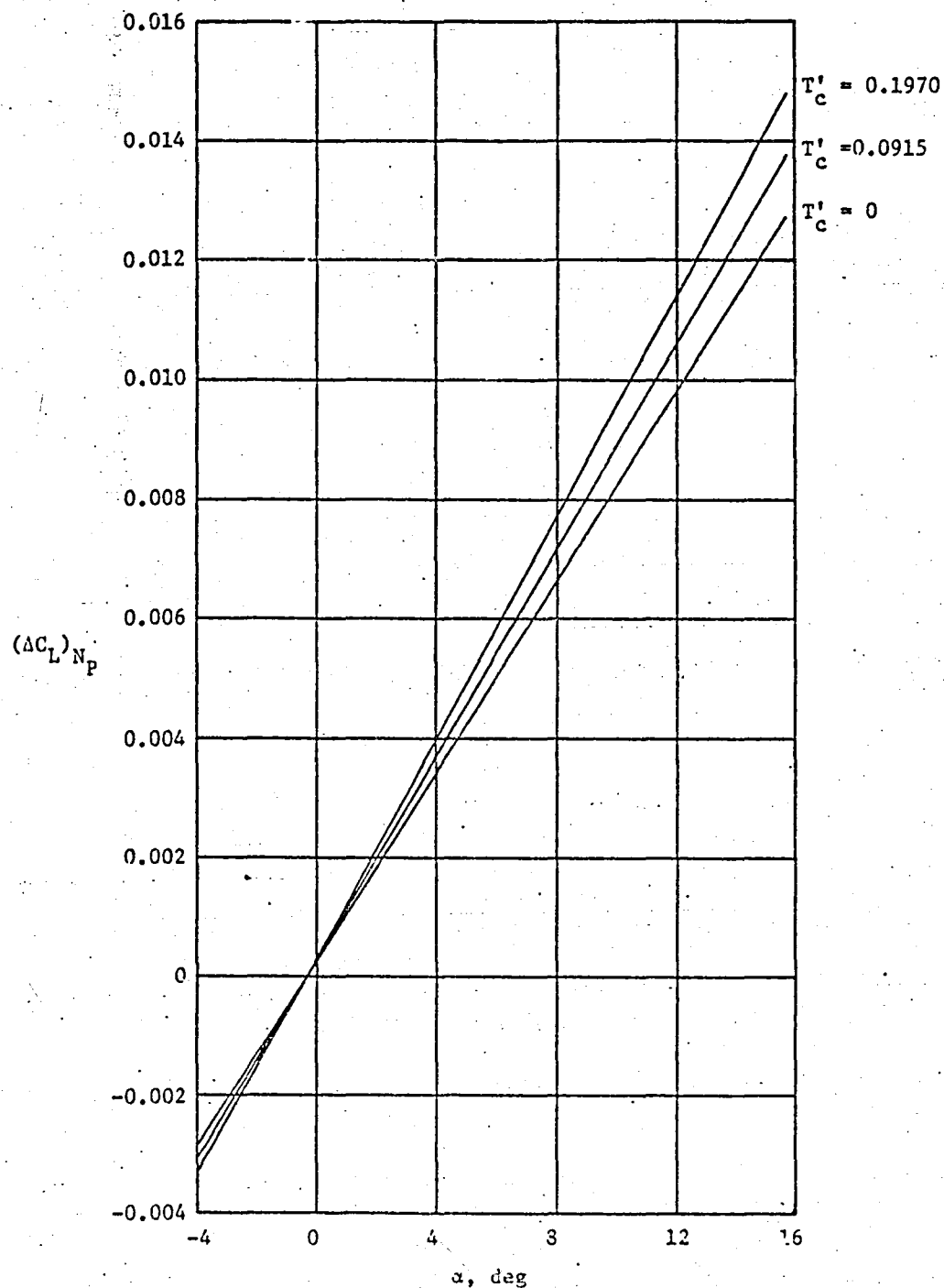


Figure 6:2.1.1: Increment in lift due to propeller normal force (Reference 1)

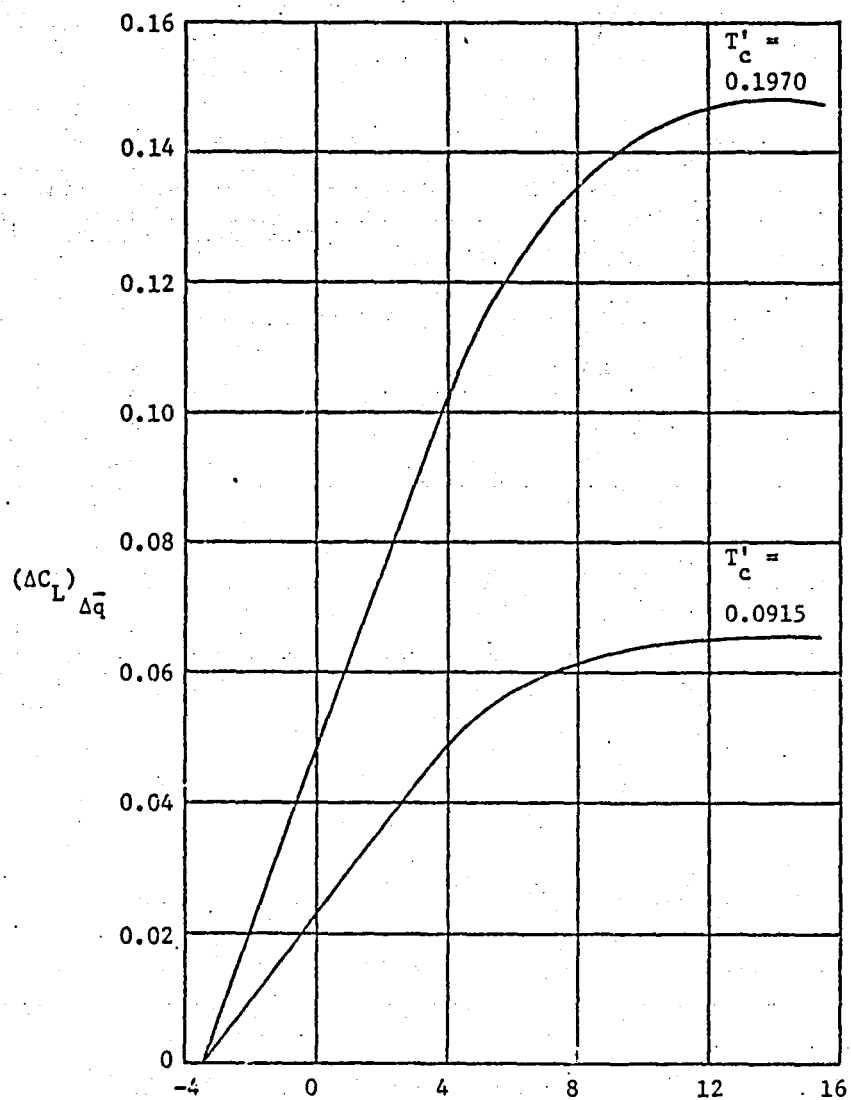


Figure 6.2.1.2: Increment in lift due to increased dynamic pressure behind the propeller (Reference 1)

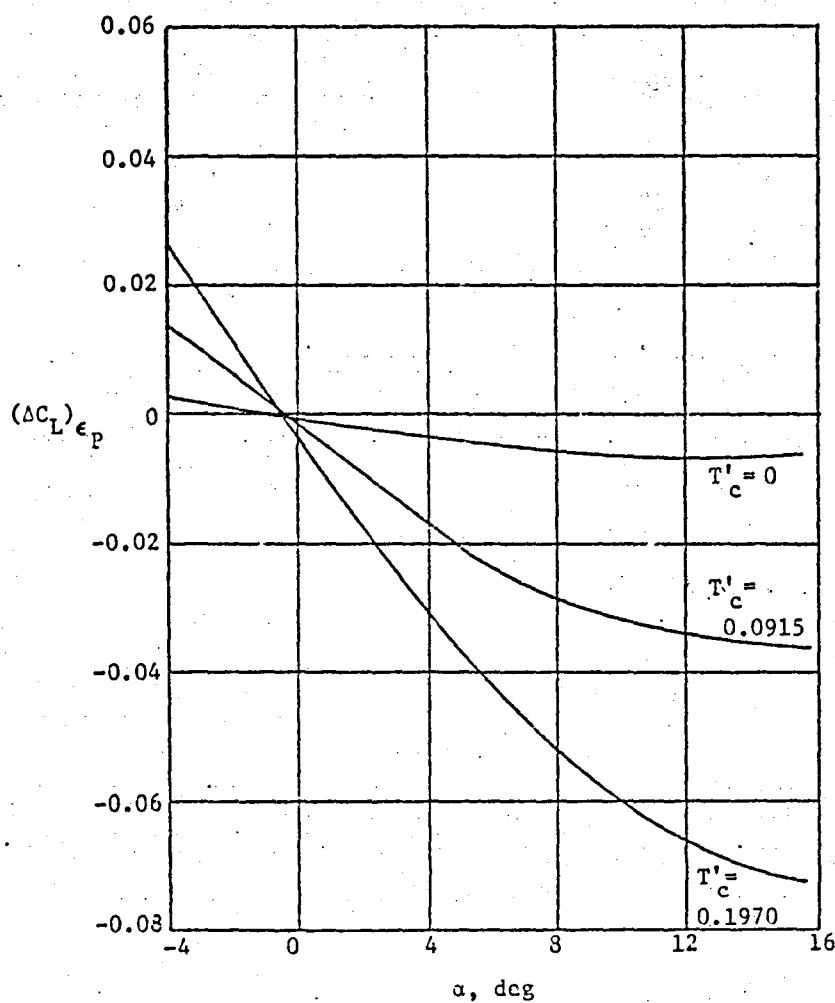


Figure 6.2.1.3: Increment in lift due to propeller downwash (Reference 1)

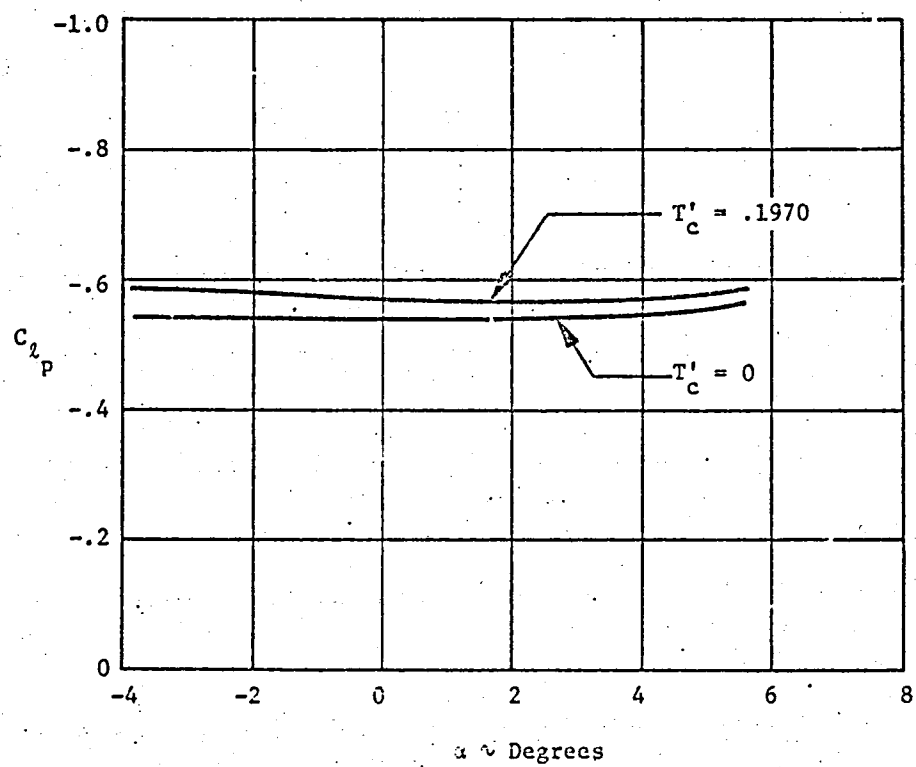


Figure 6.2.1.4: Effect of power on predicted  $C_{2p}$  of the ATLIT Airplane

## CHAPTER 7

### CONCLUSIONS AND RECOMMENDATIONS

This report presents an analytical method for predicting the lateral-directional aerodynamic characteristics of light twin-engine airplanes. This method is applied to the Advanced Technology Light Twin (ATLIT) airplane, and the predictions are then compared to full-scale wind tunnel data.

Based upon this comparison, the following observations and conclusions are presented:

1. The predicted values of  $C_{Y_B}$  are lower than the wind tunnel values; however, the overall value is fairly reasonable, especially for preliminary design. It is necessary to consider more completely the effect of angle of attack. The two areas that are thought to need the most study are the fuselage  $C_{Y_B}$  variation with angle of attack and the fuselage-vertical tail interference variation with angle of attack.
2. For the power-off case at least, the average predicted values of  $C_{n_B}$  show fairly good agreement with wind tunnel data. The tunnel tests show large variations of  $C_{n_B}$  with angle of attack, especially with power on. The contribution of the fuselage is strongly influenced by the wing location, and only partial methods are available to account for it. A more detailed study needs to be conducted to determine the effect of wing location and angle of attack.

3. The predicted  $C_{L3}$  of the ATLIT is slightly higher than wind tunnel results but still compares quite well.
4. The spoiler performance, both rolling and yawing moments, is predicted somewhat higher than found in the wind tunnel data. However, the agreement is still quite good considering the highly general nature of the methods used. A method should be developed which is tailored to general aviation airplanes.
5. The prediction of the rudder derivatives was disappointing. But again it appears that the angle of attack factors have not been accounted for. The angle of attack variation of the wing and body wake needs to be included in the prediction method.
6. The dynamic derivatives are also presented in this report. When experimental data becomes available, it should be correlated with these predictions to evaluate the prediction method.
7. There needs to be more power-on wind tunnel data in order to better correlate the effects of power. Only limited power data was available, and it was at a relatively low thrust coefficient.
8. The data in this report and in Reference 1 should be correlated with flight test data as it becomes available.

#### REFERENCES

1. Roskam, J.; Van Dam, C., and Griswold, M., "Comparison of Theoretically Predicted Longitudinal Aerodynamic Characteristics with Full-Scale Wind Tunnel Data on the ATLIT Airplane," KU-FRL-399-1, University of Kansas, July 1979.
2. Hassell, J. L., Jr., "Full-Scale Wind Tunnel Investigation of an Advanced Technology Light Twin-Engine Airplane (ATLIT)," Proposed NASA TP, 1979.
3. Wolowicz, C. H., and Yancey, R. B., "Lateral-Directional Aerodynamic Characteristics of Light, Twin-Engine Propeller Driven Airplanes," NASA TN D-6146, October 1972.
4. Finck, R. D., and Hoak, D. E., "USAF Stability and Control DATCOM," Air Force Flight Dynamics Laboratory, Wright Patterson Air Force Base, Ohio, October 1960 (Revised 1975).
5. Holmes, B. J., "Flight Evaluation of an Advanced Technology Light Twin-Engine Airplane (ATLIT)," NASA CR-2832, July 1977.
6. Wolowicz, C. H., and Yancey, R. B., "Longitudinal Aerodynamic Characteristics of Light, Twin-Engine, Propeller Driven Airplanes," NASA TN-D-6800, June 1972.



APPENDIX A

A SIMPLE METHOD FOR COMPUTING

WING VISCOUS DRAG,  $C_{D_o}$

## APPENDIX A

### A SIMPLE METHOD FOR COMPUTING

#### WING VISCOUS DRAG, $C_{D_o}'$

The wing drag which varies with angle of attack is comprised of two components. First is the drag due to direct lift production (vortex drag). The second is a type of profile drag caused by the buildup of the upper surface boundary layer as angle of attack increases (viscous drag). Together these two factors comprise the induced drag. Since the viscous drag term is needed for the prediction of  $C_{n_p}$ , the following procedure from Reference 6 is given.

Equation B.1 gives the viscous drag of a wing.

$$C_{D_o}' = k \Delta \quad (A.1)$$

where

$k$  is an empirical sweep correction factor obtained from

Figure A.1 as a function of sweep and the parameter

$J$  given in Equation A.2

$\Delta$  is the empirical viscous drag increment factor from

Figure A.2, also a function of  $J$

and where

$$J = 0.3(c_1 + 1) \frac{A_w}{B_1} \cos \Lambda / e \left[ (c_1 + 1)(c_2 + 1) - \left[ \frac{(c_2 + 1) \tan \Lambda / e}{7} \right]^3 \right]^{1/4} \quad (A.2)$$

where

$c_1$  and  $c_2$  are taper ratio constants obtained from Figure A.3.

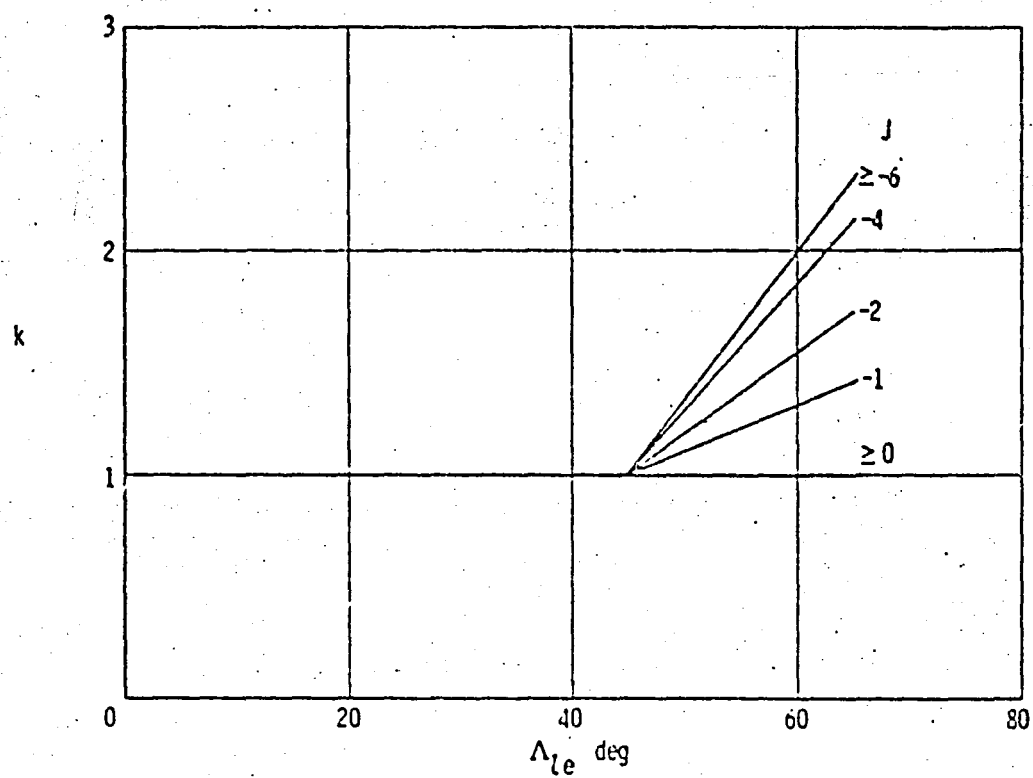


Figure A.1: Sweep correction factor (Reference 6)

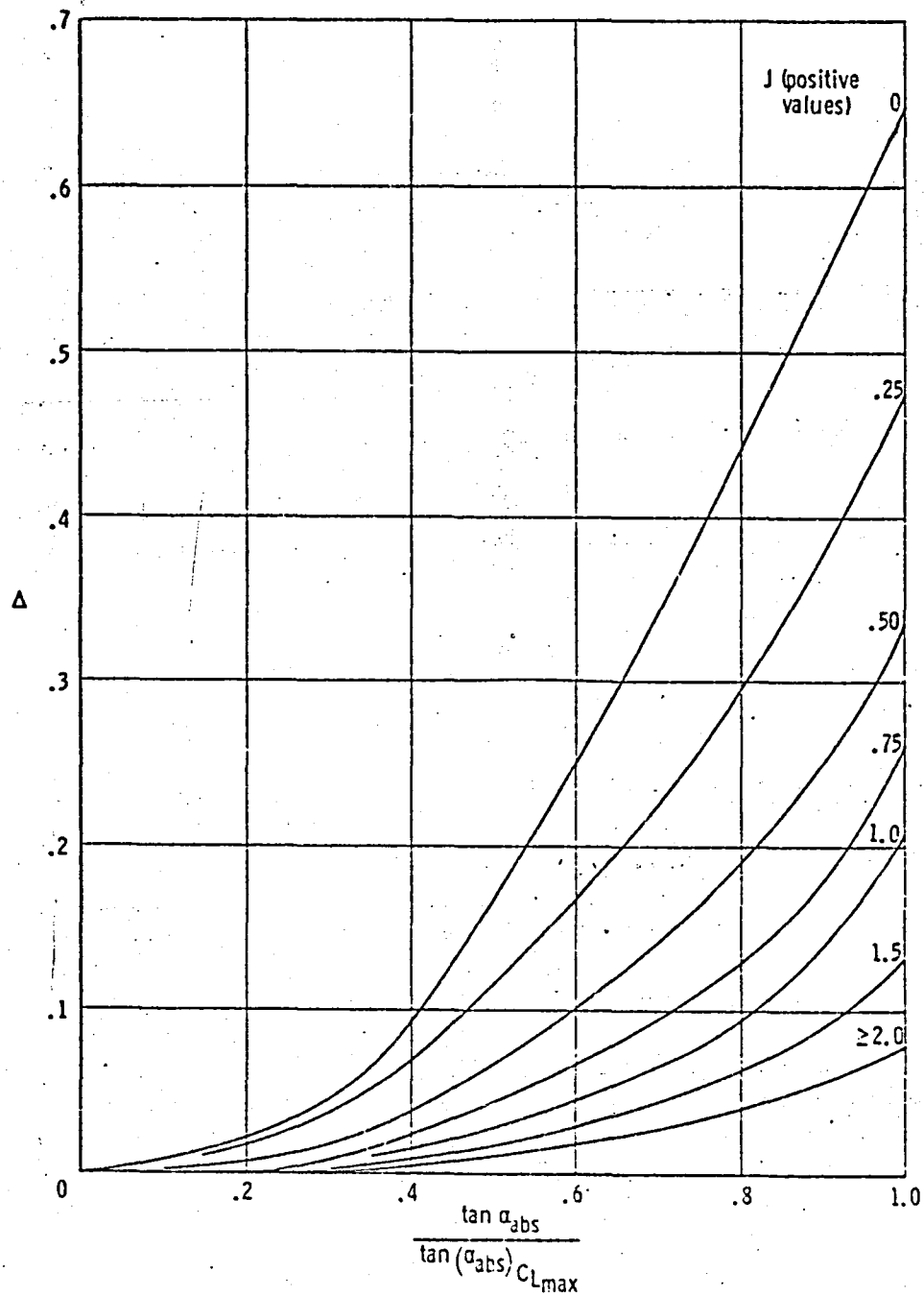


Figure A.2: Viscous drag increment factor (Reference 6)

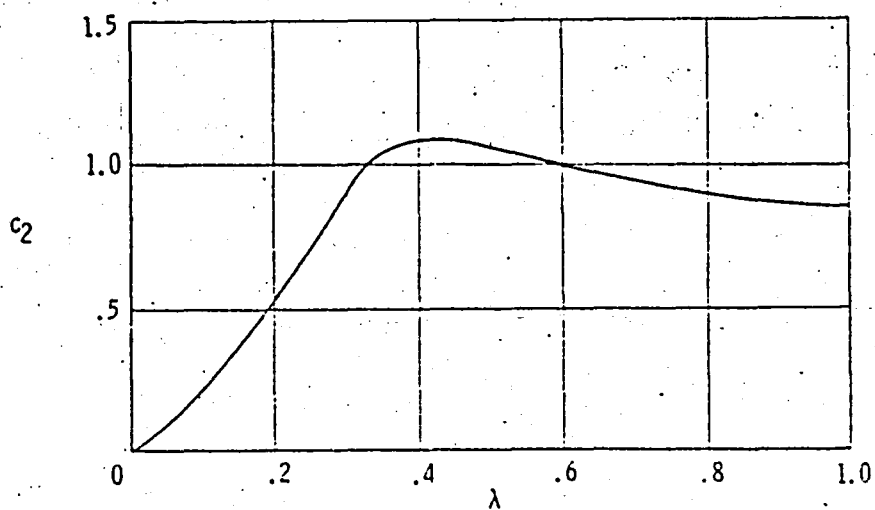
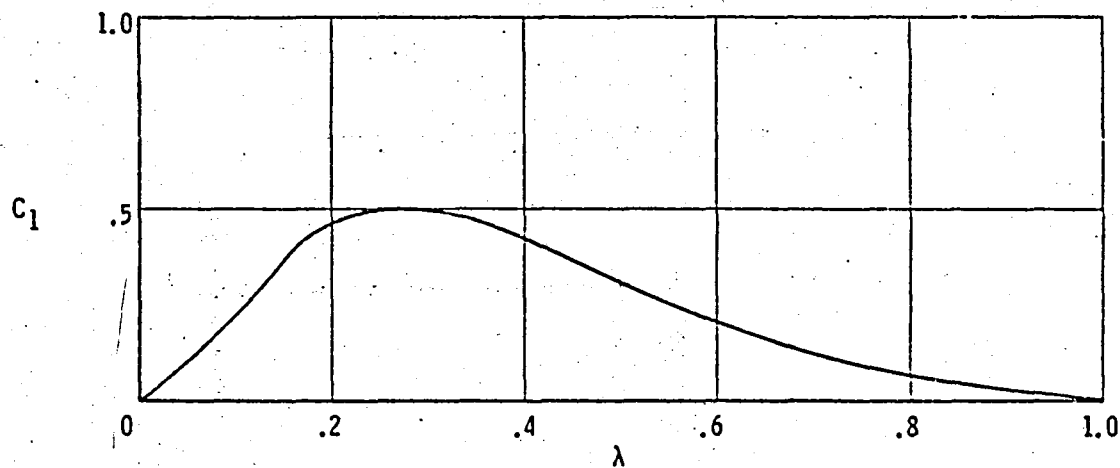


Figure A.3: Taper ratio factors for  $C_{D_o}$  (Reference 6)

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